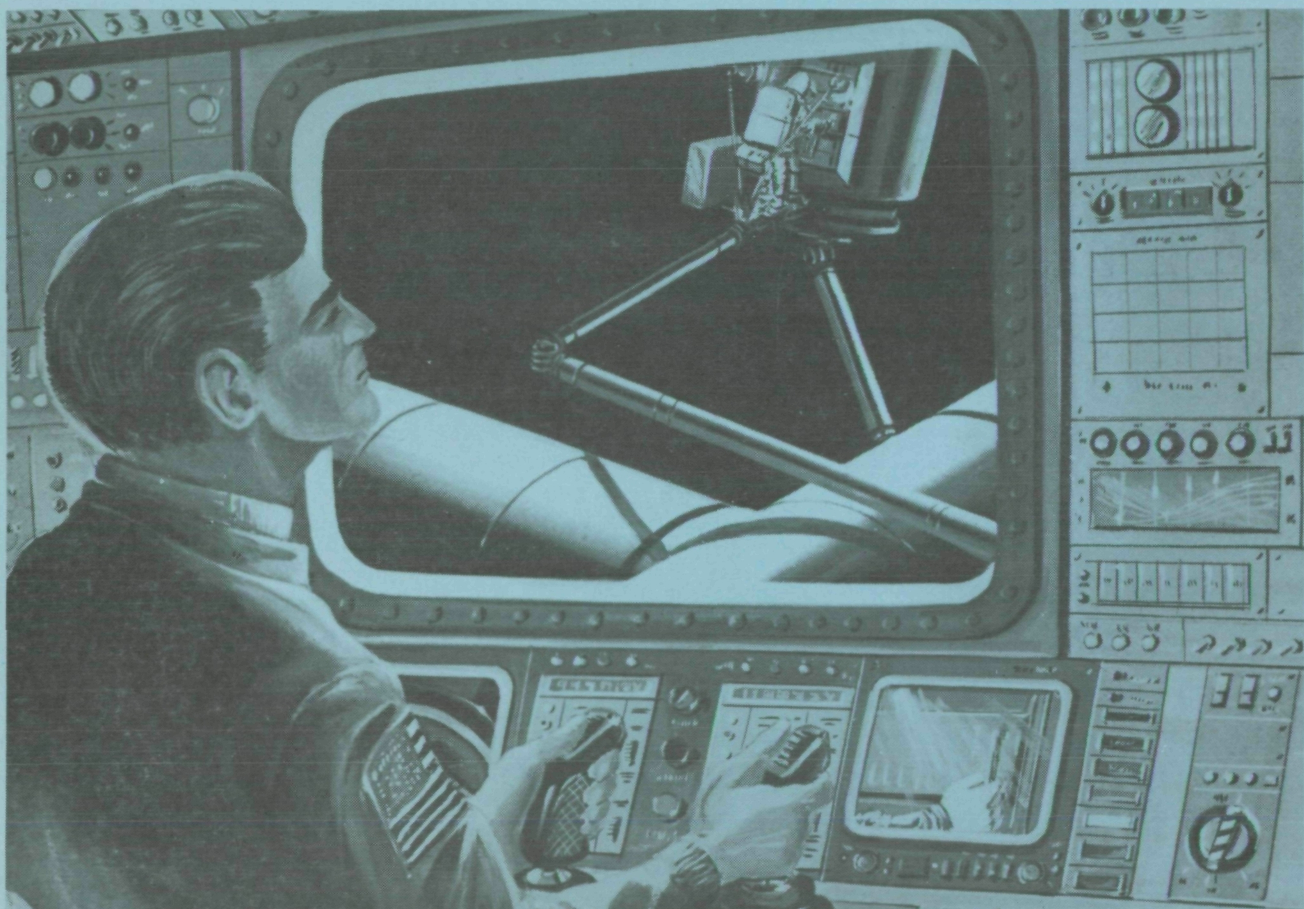
N84-18303

final technical report

SPACE STATION NEEDS, ATTRIBUTES, AND ARCHITECTURAL OPTIONS

volume II - book 2
part IV — international reports



GRUMMAN

COMSAT GENERAL

GENERAL ELECTRIC

final technical report

SPACE STATION NEEDS, ATTRIBUTES, AND ARCHITECTURAL OPTIONS

volume II - book 2

part IV — international reports

prepared for

National Aeronautics and Space Administration
Headquarters
Washington, D.C. 20546

under contract NASW-3685
Space Station Task Force
Contracting Study Project Manager — E. Brian Pritchard

by

Grumman Aerospace Corporation
Bethpage, New York 11714

report no. SA-SSP-RP008

20 April 1983

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ERNO RAUMFAHRTTECHNIK GMBH	1 Telex ☐ Datafax ☒	2 No. 3271 Date: 4.11.1982 / se

3

To: Grumman Aerospace Co., Bethpage, New York
Mr. Dick Kline, Director Shuttle Applications

From ERNO: Peter Kunigk, RV221

Info: Dr. Pollvogt, ERNO USA
W. Wienss, PRV2, U. Riedel, TEEX2

Subject: Your Questionnair Related to European Participation in US Manned Space Station

Your questionnair has been subdivided in three sections:

1. Space Station International Missions
2. Spacelab Follow-on Questions
3. EURECA Questions.

Attached you will find our inputs for the third section. For the first and second section our papers are in preparation and we intend to send them per datafax on Friday, Nov. 5, 1982 or on Monday, Nov. 8, latest.

Kind regards,

P. Kunigk W. Wienss

1. Introduction

EURECA being transported to and from orbit by the Space Shuttle and staying there actively on its own for a time period of about 6 months will provide a capability to gain more experience in the area of microgravity experiments, specifically for material processing and life sciences payloads, during this extended mission duration.

Two concepts for free-flying retrievable carriers have been investigated during prephase A study. The two concepts are based on the Spacelab Pallet on the one side and on the modular building block SPAS (Shuttle Pallet System) on the other side. These two concepts are shown in the Fig. 1.

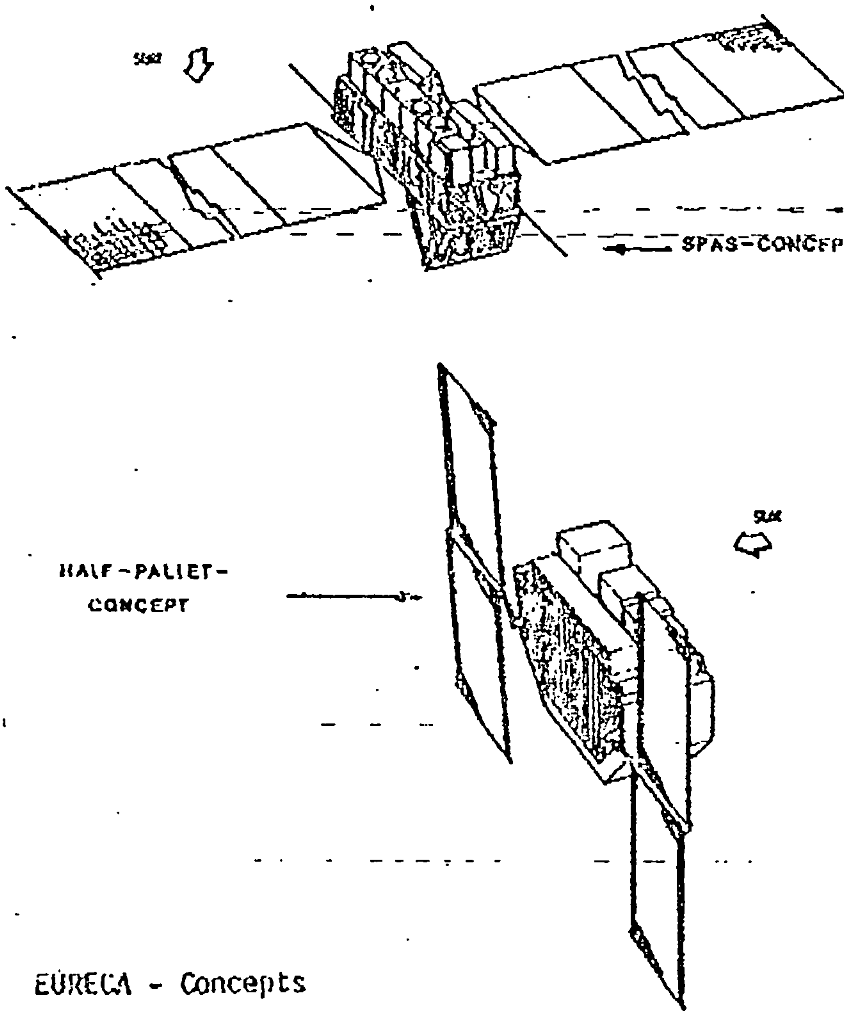

Fig. 1 : EURECA - Concepts

2. Main Characteristics of EURECA

Services to P/L :

o	P/L Mass	~1100 kg	
o	Power to P/L	1.7 KW	
o	P/L Thermal Control	1.7 KW	active cooling loop and passive thermal control
o	Data Handling for P/L	2.5 k-bits per second	
o	Microgravity Level for P/L	$<10^{-5}$ g	design requirements

Carrier Characteristics :

o	Launch Mass	~ 3500 kg
o	Carrier Length	~ 2400mm
o	Mission Duration	~8 months
o	Operational Mission Phase	6 months
o	Dormant Phase	max.2 months
o	Deployment/Retrieval Orbit	300 km; i= 28.5°
o	Orbit Control Capability	- boostup to 480 km - boost-down to 300 km - $\Delta\Omega$=1 ° plane adjustment - 360° phasing capability

3. Accommodation of Payloads

In a first step 10 experiments out of 3 different categories were selected as the reference payload and they were accommodated on a half pallet structure for the first concept (Pallet-concept) and on a two segment SPAS-structure for the second concept (SPAS-concept). The accommodation for these two concepts is shown in Figures 2 and 3.

In a second step 15 similar experiments are to be accommodated on slightly modified structural configurations. These investigations are being carried out presently.

Fig. 2 shows the reference payload accommodation for the SPAS-concept and Fig. 3 presents the same reference payload accommodated on the Pallet-concept.

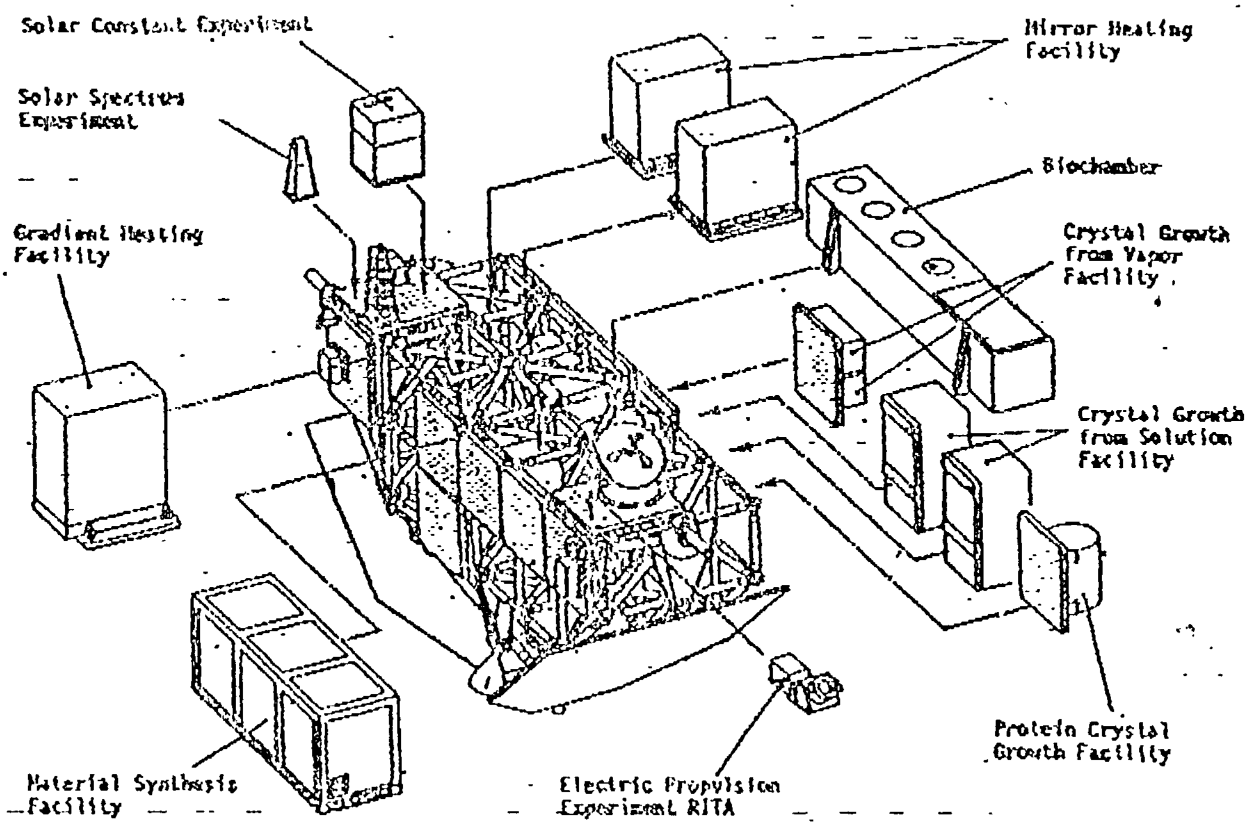

Fig. 2 : Reference Payload Accommodation (SPAS-Concept)

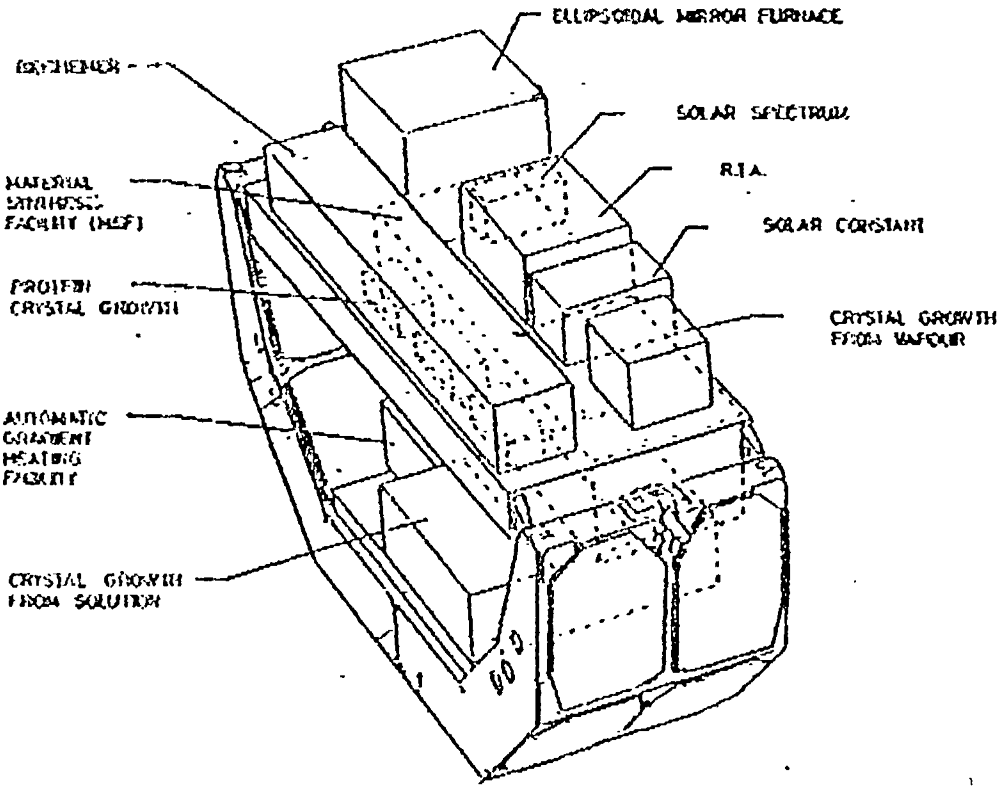

Fig. 3 : Reference Payload Accommodation (Pallet-Concept)

4. Cooling

For the Thermal Control (TC) System several possiblities combining passive and active cooling methods including heat pipes have been investigated. The preferred overall TC concept is outlined in Fig. 4 and advantages and disadvantages are compared.

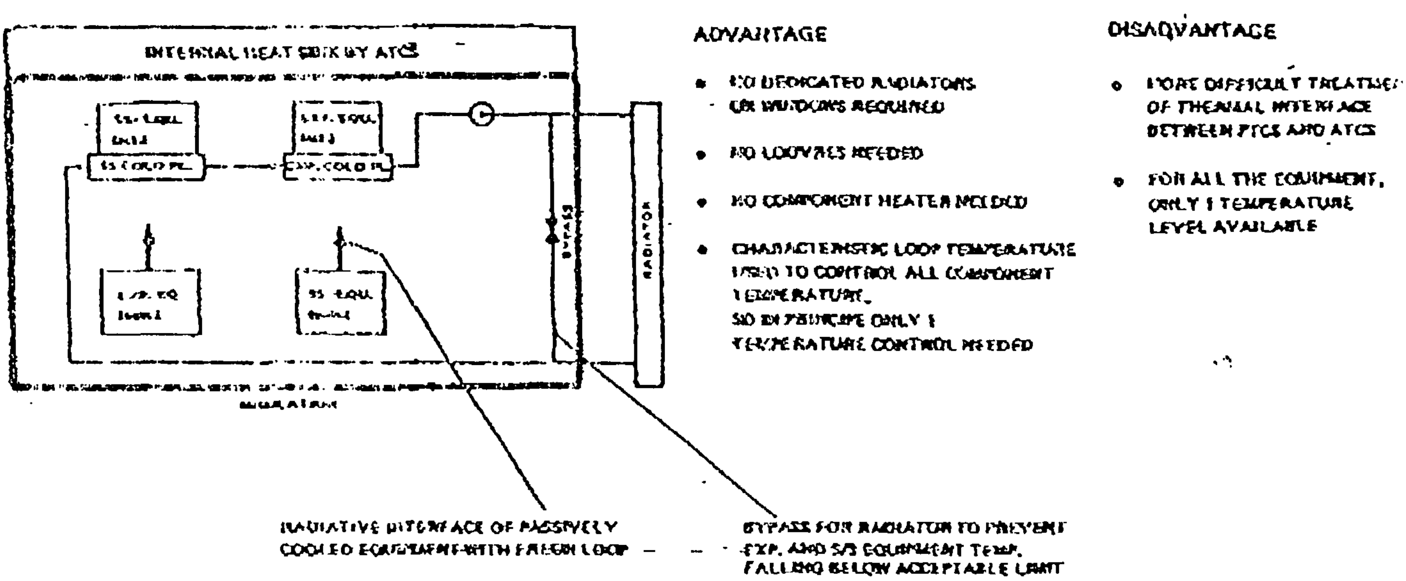

Fig. 4 : Thermal Control System Preferred Overall Concept

The evaluation and comparison between the active alternatives are continuing and are treated in considering the needs of the specific payloads without restricting the thermal subsystem to one specific payload constellation.

5. Data Handling

The Data Handling System (DHS) and the Telemetry/Telecommand (TTC) System are similar for both concepts. Ground contact for the S-band communication has been investigated for 250 kb/s during the operational phase. The results are summarized in Fig. 5. Also in this area investigations are continuing.

o	ESA DESIGN GOAL	:	1÷2 DATA DUMPS PER DAY		
o	PRESENT DATA RATE (PAYLOAD + CARRIER)	:	3 KB/S ≙ 10,8 MB/H	→	260 MB IN 24 HOURS, (WITHOUT MARGIN)
o	ESA – TM/TC LINK CAPABILITY (S – BAND)	:	250 KB/S MAX RATE; ≙ 15 MB/MIN	→	18 MIN CONTACT TIME REQUIRED
o	AVERAGE MAX. CONTACT TIME OF ESA – GROUND STATION	:	< 14 MINUTES		

– MORE THAN ONE GROUND STATION IS TO BE INVOLVED FOR 1 SHIFT OPERATION

– REQUIRED MEMORY CAPACITY IS 260 MB

Fig. 5 Ground contacts

6. Mission Requirements Summary

The following requirements and constraints have been considered for EURECA mission:

- o First Launch date during 1987
- o EURECA will be launched by Space Shuttle (STS)
- o EURECA delivered at the standard STS orbit (300 km circular, 28.5° inclination)
- o ascent of EURECA up to its operational altitude (approx. 500 km resulting from the decay analysis)
- o operating in a microgravity environment ($\leq 10^{-5}$ g) during 6 months
- o coasting max. 2 months in a dormant mode after the operational phase
- o descent to the STS retrieval altitude (336 km) with orbit plane adjustment and phasing manoeuvres
- o homing and retrieval by STS

7. The g-Level as a Function of Altitude

The g-level due to aerodynamic drag has been calculated, assuming the following physical data of EURECA

- o max. frontal area: $A_{max} = 70\ m^2$
- o mass: $m = 3500$ kg
- o drag coefficient: $C_D = 2.2$

Highest air density values occur at around 14 h, but considering that the solar panels are constantly oriented towards the sun, the maximum drag acceleration is expected to occur at about 18 h (air density at 18 h greater than at 6 h).

For this case, the g-levels were calculated as a function of orbit altitude and plotted in Fig. 6 . It can be seen that the g-level is always less than the limit of 10^{-5} at altitudes greater than 300 km.

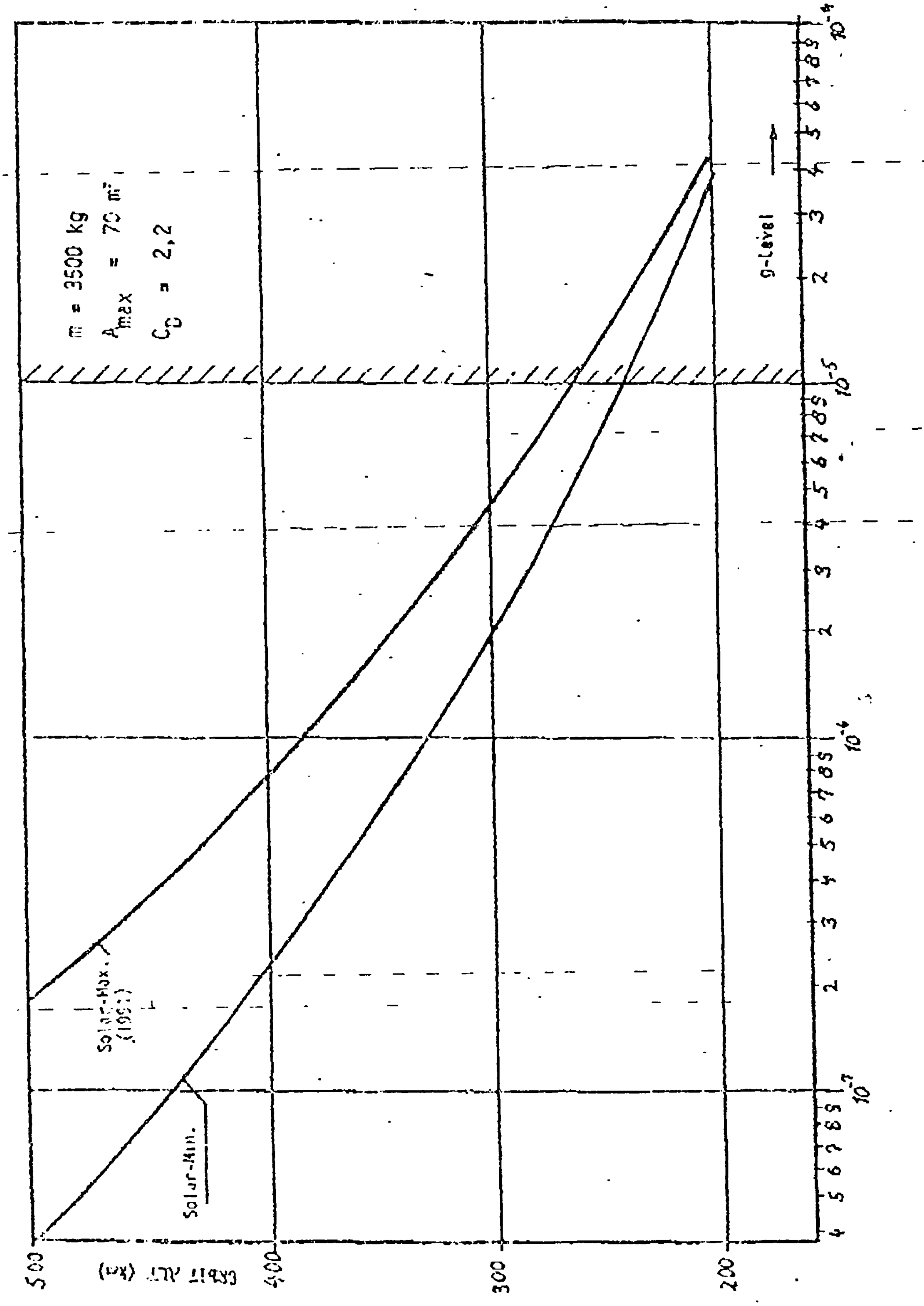

Fig. 6 : g-Levels as a Function of Orbit Altitude

11/2/IV

ERNO RAUMFAHRTTECHNIK GMBH	1 Telex ☐ Datafax ☒	2 No.: 3305 Date: 5.11.1982 / se

3

To: Grumman Aerospace Co., Bethpage 8. Nov. 1982
Mr. Dick Kline, Director Shuttle Applications

From ERNO: Peter Kunigk, RV221

Info: Dr. Pollvogt, ERNO USA
W. Wienss, PRV2, U. Riedel, TEEX2

Subject: Your questionnaire Related to European Participation in US Manned Space Station

Ref.: Our Datafax Nr. 3271 dated Nov. 4, 1982

-- Attached you will find our inputs for your "Spacelab Follow-on Questions".

In case you need any explanations or for further questions please contact Peter W. Kunigk at (421) 539 4245 oder Wolfgang Wienss at (421) 539 4449 of ERNO.

Kind regards,

P. Kunigk   W. Wienss

1. A preliminary structure strength and stability assessment of a 4-segment Spacelab module, which can be used as command Center/Habitation Module in the frame of a Space Operations Center (SOC) and which shall be transported by the Shuttle Orbiter, has been performed. The total mass (beside structure and equipment payload) was varying between 11 t and 22 t. The investigations were limited to fittings and fitting attachment areas of the module assuming a statically determined suspension.

The most important results of the analysis are:

o For the configuration of 11 t no design changes will be necessary with regard to the effect of the fitting reaction forces

o Configurations with increased total mass need different design and/or material changes for some structure parts (fittings, frames, longerons etc.).

A 3-segment module is a step between a Spacelab Long Module which is qualified and a 4-segment module. That means, the conditions for a 3-segment module are correspondingly more favourable as for a 4-segment module.

2. Assuming a step-by-step-evolution of the Spacelab module, the availability of such a module, which fulfils the known specifications for SOC/SAMSP, is expected for use in the frame of a manned Space Station in approximately 1994.

3. Spacelab modules can be supplied for later inclusion of subsystems and other mission related equipments for use as building block for a Space Station.

If it would be required to increase the Spacelab module skin thickness (due to radiation/meteorite, space débris) in order to achieve an on-orbit life time of 10 years with on-orbit maintenance, a change in the manufacturing process will be necessary. The skin thickness could be increased due to the fact that the cylinder sections are milled down to a 1.6 mm thickness using a 20 mm material.

Cost and Schedule: TBD

4. Details of Subsystems

4.1 Igloo Subsystems have not been analysed in detail for possible use in a Space Station, however, in general it can be stated that Igloo equipment to be used in habitable areas of a Space Station can be considered being qualified in principle. This because most of the equipment is also applied in the Spacelab module. Where Igloo equipment is intended to be applied for Space Station in non-habitable areas, it is the intent not to offer an Igloo type housing, but to qualify this equipment to be space hardened.

4.2 Module Subsystems

The following reflects an extract of results obtained during a Spacelab FOD-Medium Term Study, performed in 1981/82, concerning the use of modules/subsystems in the frame of a U.S. manned Space Station.

4.2.1 Environmental Control and Life Support

The basic system configurations for the ECLS analysis have been defined in a manner that a maximum overall configuration commonality for various platform applications has been achieved:

- Experiment Module (EM) in 2 segments,
- Habitat Module (HM) in 2 segments, and
- Habitat Module (HM) in 4 segments.

Every module shall form an integral part of the relevant manned space platform (SOC or SAHSP). The crew size per module is 4 men and the mission duration 90 days.

The major differences from the Spacelab performance requirements are the reduced total pressure from 1.013 bar (14.7 psia) to 0.813 bar (11.8 psia) to avoid EVA pre-breathe, and reduced carbon dioxide partial pressure to max. 3.8 mm Hg. The applied ECLS design loads represent the standards for manned space flights for longer mission durations.

The design-driving safety criteria are the philosophy of a "safe-haven", and the permitted closed hatch operation for the EM. The required amount of consumables for a 4 man crew on a 90-day mission is considerable (758 kg of atmosphere constituencies and 7348 kg of water). This makes the use of regenerative concepts for the major ECLS functions in particular CO_2 removal, water reclamation and oxygen and nitrogen supply imperative. Optional processes for the various ECLS functions have been analysed as to the interaction with each other in various ECLS concepts and in order to achieve maximum weight, volume and power savings.

Experiment Module

The Experiment Module (EM) is first envisaged in a platform attached, Shuttle tended mode, only manned for 7-20 days during Shuttle revisits. For the rest of the on-orbit stay time the equipment of the module shall work automatically.

The first improvement from the Spacelab configuration is most likely the incorporation of a regenerative carbon dioxide removal system (Fig. 4.2-1). The weight of the LiOH concept for a 4 man crew during a 20 day tended operation is 147 kg at 3.8 mm Hg CO_2 without the fixed hardware weight and additional storage. The additional weight of a solid amine system with steam regeneration (without redundancy as Shuttle is the safe-haven) is 119.3 kg.

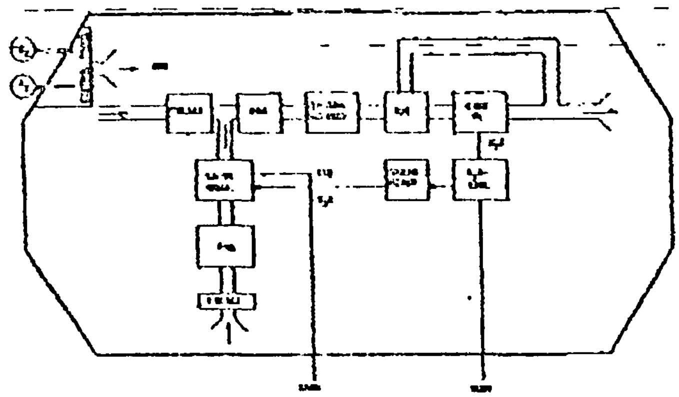

Fig. 4.2-1: ECLS Schematic for Experiment Module, Shuttle Tended Mode

Furthermore, if a certain carbon dioxide control is required during the unmanned periods a SAWD* system is inevitable as it can perform automatically. Furthermore this improvement would be a step towards the manned EM for long term missions.

The EM attached to a permanently manned space platform (SOC or SAMSP) is very closely involved in the overall regenerative concept (Fig. 4.2-2) as any metabolic products not reclaimed in the EM but wasted in an expendable system must be re-supplied by the Shuttle no matter what regenerable systems are incorporated into the HM. Only regenerative carbon dioxide removal and water recovery concepts are required for a none-habitable module. The weight savings for one 90 day mission are more than 1242 kg for the EM-concept in Fig. 4.2-2.

Habitat Module

The design of ~~an ECLS system for the~~ HM is very strongly governed by the safety criteria.

The ECLS SAMSP configuration selected (Fig. 4.2-2) for a 4 man crew per module has the following design features:

- High degree of commonality as the individual functional components (CO_2-removal, H/X, fans, water tanks etc.) can be design for the same performance characteristics.
- The fluid connections at the docking/berthing adapters are limited to water, oxygen, nitrogen and carbon dioxide. No hydrogen connections are needed between the modules.

*SAWD = Solid Amine Water Desorbed

- The safety criteria are in general met for this particular configuration of modules.
- The weight advantage for the HM is 6843 kg for the first 90 days.

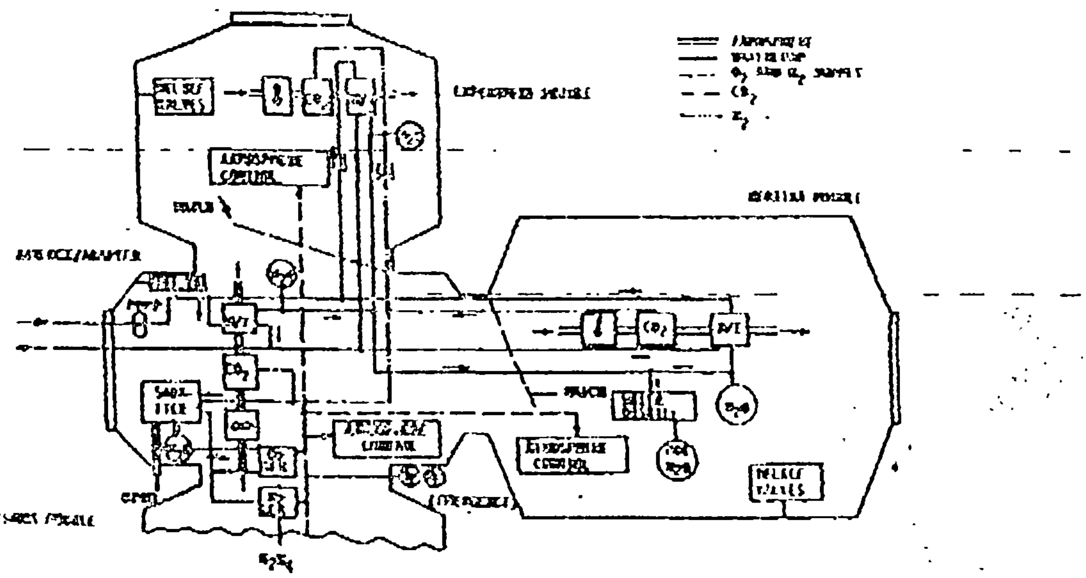

Fig. 4.2-2: ECLS SAMSP Habitat and Experiment Modules

For the ECLS SOC configuration the concept is similar to that for SAMSP but due to the crew size (8 men/module at emergency) the air revitalization loop in the HM must be doubled. The weight savings for a regenerable ECLS system for SOC are more than 6827 kg for the first visit, and for the resupply every 90 days over 8436 kg.

Whether a platform of the SOC or SAMSP configuration is selected, the crew size of 4 or 8 men enables identical air revitalization, water reclamation, carbon dioxide reduction and atmosphere supply loops in single or redundant arrangements for all modules involved (HM, EM and Airlock/Adapter) starting with the first Spacelab improvements for Experiment Modules. The present Spacelab is adaptable to EM functions with minor modifications, and becomes the logical building block in the evolution towards the closed loop systems.

4.2.2 Electrical Power System (EPS)

An EPS for a Habitation Module or an Experiment Module of a SOC or SAMSP type mission will contain the power

- conditioning
- fusing and protection
- switching and
- distribution

for crew, subsystem or experimenter needs.

The power will be derived from an external power source system which will consist of a solar generator and necessary batteries, forming the primary energy for the to be docked modules.

Preliminary interface power busses are expected as follows:

- 28 VDC bus
- 120 VDC bus
- 115/200 VAC bus
- 28 VDC essential bus

The baseline Spacelab module EPDS ~~can accommodate power levels~~ up to 12 kW with relatively minor modi~~fications to some equipment~~.

Suitable candidates for this task are the Experiment Power Distribution Box (EPDB) the ESA Junction Box (EJB) and the ~~S/L 400 Hz Inverter since~~ these are really multi purpose devices. They are able to handle a high level of power and provide the remote control capability. The Experiment Power Switching Panel (EPSP) will be a further candidate. The Power Control Box (PCB) and the Emergency Box (EB) can serve for primary distribution.

For the Experiment Modules the following power consumption is expected:

Material science paylaod:	6 to 10 kW
Life science payload:	8 to 11 kW

4.2.3 Avionics

The concept of U.S. manned platform with redundant module should allow to use Spacelab CDMS as concept for data handling and command distribution of the module.

The interfaces of the CDMS to the main data management subsystem shall be similar to those defined in Space Shuttle System Payload Accommodation Handbook, JSC 07700, Vol. XIV. The difference with the S/L CDMS interface will be found on Payload Data Interleaver (PDI) interface instead of PCMMU.

The MDM link shall be used for command uplink, PDI interface for telemetry downlink, and Ku-SP for high rate data coming from the CDMS multiplexer. The 1 024 kHz and GMT signals will give timing inputs to the CDMS.

The modular concept of data processing assembly of CDMS allows to accommodate requirements as:

- minimize risk of loss of modules
- selected number of on-orbit spares for in flight maintenance
- built in test capability to facilitate detection and reporting of functional discrepancies.

The power requirements will be identical to Spacelab. The CDMS will be power supplied from the EPDS with 28 VDC regulated, and 400 Hz 3 phases devoted to data display system.

The Input/Output Unit will include PDI coupler (and redundant) to ensure telemetry downlink to the communication subsystem of the station. This coupler will be a standard microprogrammed I/O Unit coupler, used to perform the telemetry formatting function. The interface adaptation including BI Ø-L is performed in a small PC board. The 64 kbps (or submultiple) is received from the 1 024 kHz MTU coupler and divided in a programmable counter (4 rates possible 64 kHz, 32 kHz, 16 kHz and 8 kHz).
The telemetry format is cyclicaly transmitted each 1 second.

The data busses (S/S and EXP) will accommodate Processor Interface Adapter (PIA) to control Processors as ECLS processor, EPDS processor, Waste management processors, etc. .. and experiment processors (DEP's). In order to control those processors the DEP protocol shall be included in SCOS (already provided by ECOS).

The only modification in the hardware/software interface is the PDI coupler instead of the PCMMU coupler.

The difference is at the level of the fetch commands. They are received by the PCMMU coupler which is then driven externally and in the PDI coupler the fetch commands are present directly in the coupler.

4.3 Pallet Subsystems

The study effort in the FOD Medium Term Study as mentioned in section 4.2 covers with regard to pallet use for space stations two principal types of application. These are the use as a payload/experiement carrier attached to the space station and the use as a free-flying carrier for different types of missions, e.g. logistics carrier, experiment carrier - see material supplied in response to your question concerning EURECA -.

With regard to the application as payload carrier permanently attached to the core station, several detailed conceptional, operational and functional aspects have been investigated. Due to the extensive study results available we have difficulties to make the proper choice, therefore, you are kindly requested to specify your questions in more detail.

5. In the frame of the just starting ESA-Study "Participation of European Industry in NASA Space Station Studies" conceptional investigations of the following most favourable candidates (from European side) will be performed:

Space Station Elements

Habitation Module
Experiment Module
Experiment Carrier
Logistics Module
Logistics Carrier
OTV-Hangar

Service and Maintenance System

Operation Support System
Teleoperator Manoeuvering System (TMS)
Service Carrier

Transport Elements

Orbital Transfer Vehicle (OTV)

Network System

Relay Satellites
Ground Station Tracking Network

Special Elements
Manipulator
Radiator

Preliminary results ~~are~~ expected at the begin of 1983.

DORNIER SYSTEM

Participation in
NASA
Space Station Study

TITLE:
TITEL: POINTING SYSTEMS

DOCUMENT NO.:
DOKUMENT NR: TN-SSS-DS-003

ISSUE NO.:
AUSGABE NR: 1

ISSUE DATE:
AUSGABEDATUM: 14.02.1983

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PROJEKTMANAGER

LIST OF CONTENTS

1. INTRODUCTION

Dornier looks back on 13 years of experience in the development and manufacture of stabilized gimbal systems, starting with balloon borne telescope pointing systems and maritime antenna stabilization systems. In the space field the following gimbal systems have been developed by Dornier:

- the Instrument Pointing Subsystem (IPS)
- the two axes antenna pointing mechanisms for the German MRSE and MRSE-MAS
- the Position and Hold Mount (PHM) covering phase A, B and demonstration model
- the Antenna Pointing Mechanism (APM).

A detailed system description for IPS, PHM and APM is given in section 2. The Space Station relevant payloads are summarized in section 3. The Space Station accommodation aspects are handled in section 4.

2. POINTING SYSTEM DESCRIPTION

2.1 Instrument Pointing Subsystem

2.1.1 General

Starting in 1976 DORNIER SYSTEM has been developing and manufacturing the Instrument Pointing Subsystem (IPS) under contract of the European Space Agency (ESA). In the years 1979/1980 the IPS has successfully passed an extensive qualification programme.

Due to numerous engineering changes concerning Spacelab and Orbiter interfaces the manufacturing programme for the flight unit had to be interrupted to allow for a redesign of critical components. The whole programme of manufacturing and requalification of the IPS flight unit will be completed by end of 1983.

The first mission of IPS is scheduled to take place in October 1984.

2.1.2 Technical Concept

The Instrument Pointing Subsystem (IPS) provides precision 3-axes pointing for payloads which require greater pointing accuracy and stability than is provided by the Orbiter. The IPS can accommodate a wide range of payload instruments of different sizes and weight. The overall configuration of IPS with a payload is shown in Figure 2.1.2-1.

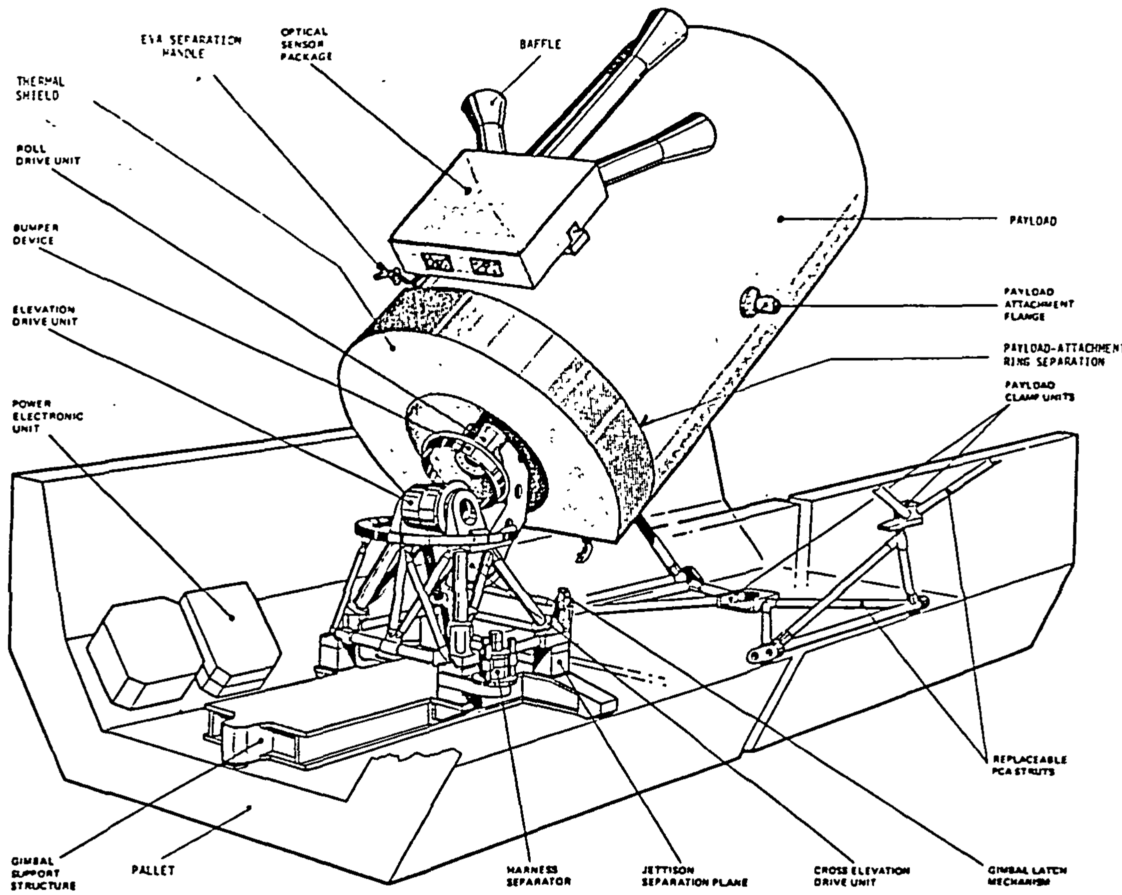

Fig. 2.1.2-1: IPS Configuration on a 2 Pallet Train

The IPS features the following main systems:

- the three axes gimbal system for precision pointing
- the Payload Clamp Assembly (PCA) to support the payload during ascent and descent
- Attitude Measurement Assembly including Optical Sensor Package and Gyro Package
- the Power Electronics Assembly (PEA)
- the Data Electronics Assembly (DEA)

For the wide range of various payloads different IPS configurations can be provided:

- Flexibility is given by potential payload dimensions from 0,5 m to 3 m diameter and up to 2 m distance from Payload CG to IPS attachment plane

- Payloads up to 3000 kg mass can be accommodated

- Single or double pallet train may be used depending on payload mass and dimensions

- The field of view is conical about the Orbiter z-axis ant its half-cone angle can be varied from 30° to 60° by adjustment of the impact ring position

- Any x position of the centre of rotation (COR) can be adapted by moving the gimbal system on the gimbal support structure rails

- The z position of the centre of rotation (COR) can be adjusted in height from 1,3 m to 2 m by means of a mission dependent replaceable column

- The payload clamp assembly can adapt the wide range of payloads by replacement of struts

- Nominal payload characteristics as used for IPS design reference are shown in Table 2.1.2-1

- The IPS electrical and mechanical data and the essential IPS capabilities for experiment accommodation are summarized in Table 2.1.2-2.

	Large Payload	Small Payload
Mass	2000 kg	200 kg
Dimensions	2 m Ø x 4 m	1 m Ø x 1,50 m
Moment of inertia about payload CG:		
about axis perp. to LOS	1460 kgm^2	20 kgm^2
about LOS axis	1000 kgm^2	25 kgm^2
CG offset referred to P/L interface plane:		
along LOS (Note 3)	1,63 m	0,50 m
perp. to LOS	0,00 m	0,10 m
characteristic structural frequency (Note 1)	7,5 Hz	(Note 2)

Note 1: Lowest bending mode, supported at P/L interface plane

Note 2: Considered not to be critical

Note 3: The LOS is the vector through the COR, perpendicular to the P/L interface plane

Table 2.1.2-1: Characteristics of Nominal 2000 kg and 200 kg Payloads

The optical sensor package includes the capability to have two roll sensors at a skewed angle of either 45 degrees or 12 degrees with respect to the line of sight (LOS) of the centrally mounted optical sensor. The LOS's of all three optical sensors are arranged in one plane. Provision is also made for the mounting of a light baffle system, designed for specific mission conditions, at the aperture of each optical sensor but structurally decoupled from the sensor.

Requirement	Value	Comments
IPS Total Mass	1174 kg	For the SL-2 configuration (1.83 m COR height); without mission dependent PCA struts, including integrated SL hardware (RAU/IS)
IPS Power Consumption		
- Inertial Pointing	500 W	Mean Value
- Full Torque Slewing	1400 W	30 Nm Torque About 2 Axes
- Emergency Stowage	200 W	Short Time Peak Value
- Thermal Control	350/1200 W	Cold Case Mean/Peak Power
Payload Accommodation		
- Nominal Mass	2000 kg	
- Max. PCA Loading Mass	3000 kg	With Stiffened Clamping Struts
- Max. P/L Pointing Mass	7000 kg	not covered by actual clamping system
- Diameter	0,5 to 3 m	
- Back to CG Offset	0,5 to 2 m	3 m for 7000 kg P/L
Payload Support Power		
- Main Power	1250 W/22 VDC	8 x 6 AWG 20 independently fused
- Essential Power	100 W	4 AWG 20
Payload Date Lines for		
- Experiment RAU's	3	Data and Power Busses
- High Data Rate	6	6 TSP 125 ohm for 16 Mb/s
- Analog Signals	3	3 TSP 75 ohm for 4,5 MHz
- Control	10	for discrete signals and commands

Table 2.1.2-2: IPS Physical Characteristics and Payload Accommodations

2.1.2.1 Mechanical Concept

The IPS comprises two major mechanical assemblies:

- the three axes Gimbal Structure Assembly
- the Payload Clamp Assembly (PCA)

2.1.2.1.1 Gimbal System

During operation on orbit the payload is attached to the Gimbal System and its three axis attitude and stability control is performed by torquers applied by the three identical drive assemblies. Their axes intersect at one point. Each drive unit employs 4 ball bearings, two brushless DC-torquers, and two single speed/multi speed resolvers for nominal and emergency operations.

The electronic units of the IPS are mounted on the Equipment Platform, except the power electronics unit which is mounted to a cold plate on the Pallet.

During launch and landing the gimbal system and the payload are separated, so that no additional loads or moments will be imposed on the payload by the gimbal structure.

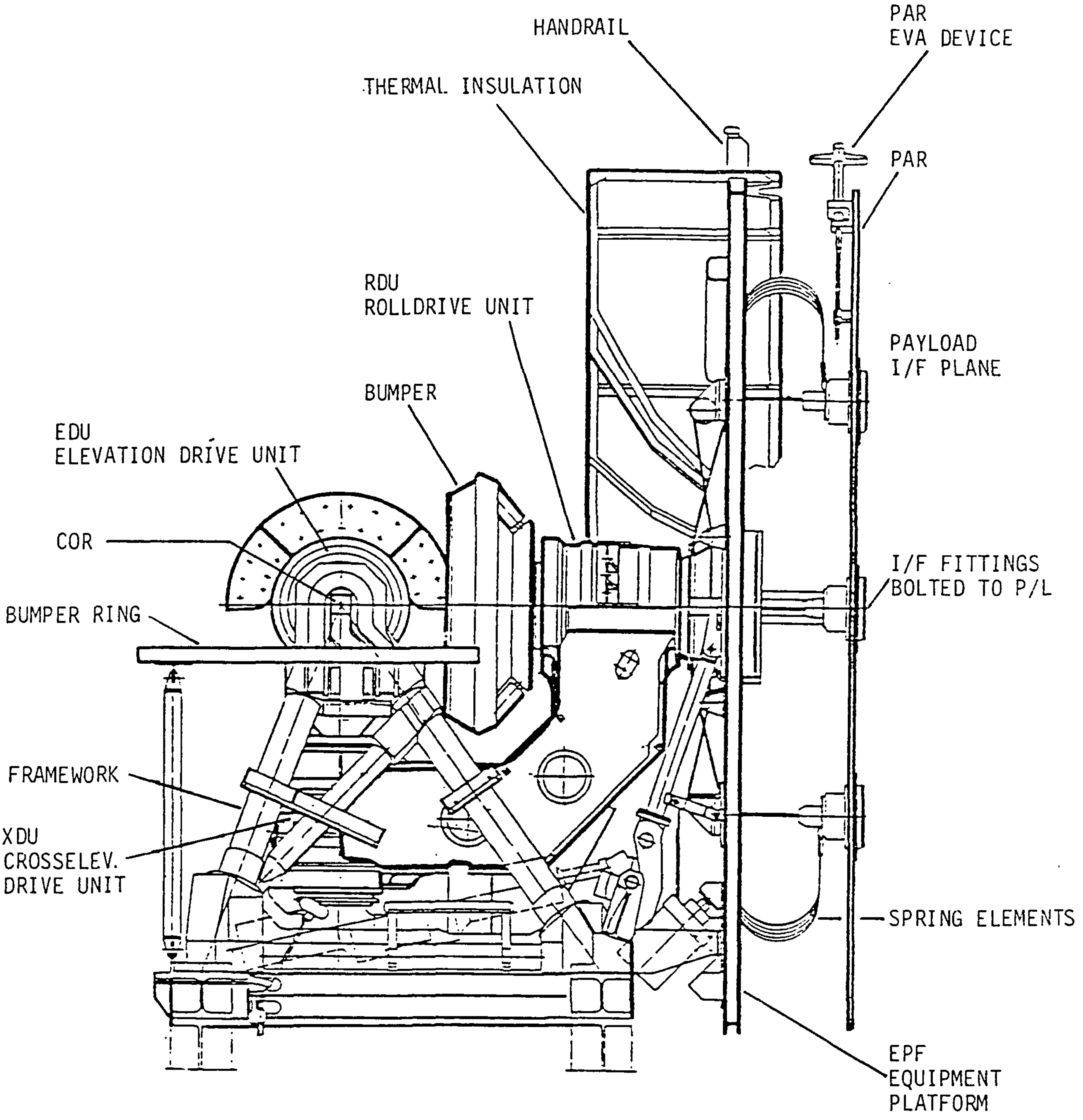

FIG. 2.1.2.1.1-1: GIMBAL STRUCTURE

The gimbal support structure, mounted at 4 hardpoints to the pallet, includes rails with brackets for positioning the gimbal system in x direction. For adjustment in x and y direction the brackets include a screwing device.

A replaceable extension column between the Gimbal Support Structure (GSS) and the jettisoning device at the Gimbal Structure can adapt the height of the COR between 1,3 m to 2 m.

The gimbal system includes a jettison device for use in an emergency case in which the payload and/or IPS cannot be retracted to a safe landing configuration and overboard jettison is required.

The payload and the integrated gimbal structure will be installed separately onto the pallet and then be connected to each other via three mounting flanges on the PAR. During the ascent/descent phase the upper gimbal structure is locked in its adjusted location by the Gimbal Latch Mechanism (GLM).

After release of the PCA clamping devices on orbit a mechanism will move the Payload Attachment Ring with the payload towards the EPF, and clamp the Payload onto the Gimbal Structure.

In an emergency case a PAR mounted EVA device is dedicated to separate the P/L, connected to the PAR, from the Gimbal Structure for individual jettison of either part.

2.1.2.1.2 Payload Clamp Assembly

During ascent and descent, the payload is physically separated from the Gimbal Structure to avoid any load path. The payload is supported by the Payload Clamp Assembly (PCA) which distributes the flight loads of the payload into the pallet hardpoints (Figure 2.1.2.1.2-1). The PCA is designed such that the directions of the loads induced in the payload are predominantly tangential.

The Payload Clamp Assembly consists of

- clamping mechanism, i.e. three clamping units (CU) defining a triangle in the Y_o - Z_o plane and an actuator mechanism with replaceable flexible shafts to drive the CU's

- replaceable struts distributing the loads from each clamping unit to four pallet hardpoints. These struts will be tailored to each individual mission configuration and thus determine the size of the triangle mentioned above to accommodate payloads between 0,5 m to 3 m diameter.

- non-replaceable elements distributing the loads from the replaceable struts to the pallet hardpoints.

- for an emergency case an EVA device is provided to enable the removal of the keybolts and so release the payload.

The Payload Clamp Assembly is capable of mounting and distributing the load of a 2000 kg payload into a single unmodified pallet without exceeding safe loading conditions on the basis of compatible payload dimensions and CG location. However, the clamping mechanisms and the non-replaceable elements of the Payload Clamp Assembly are designed for the loads correspond-

ing to a payload mass of 3000 kg with its CG located up to 5 cm radially displaced in the y-z plane from the centre of a 3 m diameter circle in the clamps and up to 10 cm displaced in the X direction from the plane of the clamps. In this case, the pallet may require local reinforcement in the location of PCA/ pallet attachment points, and the replaceable struts of the PCA must be designed to the loads involved.

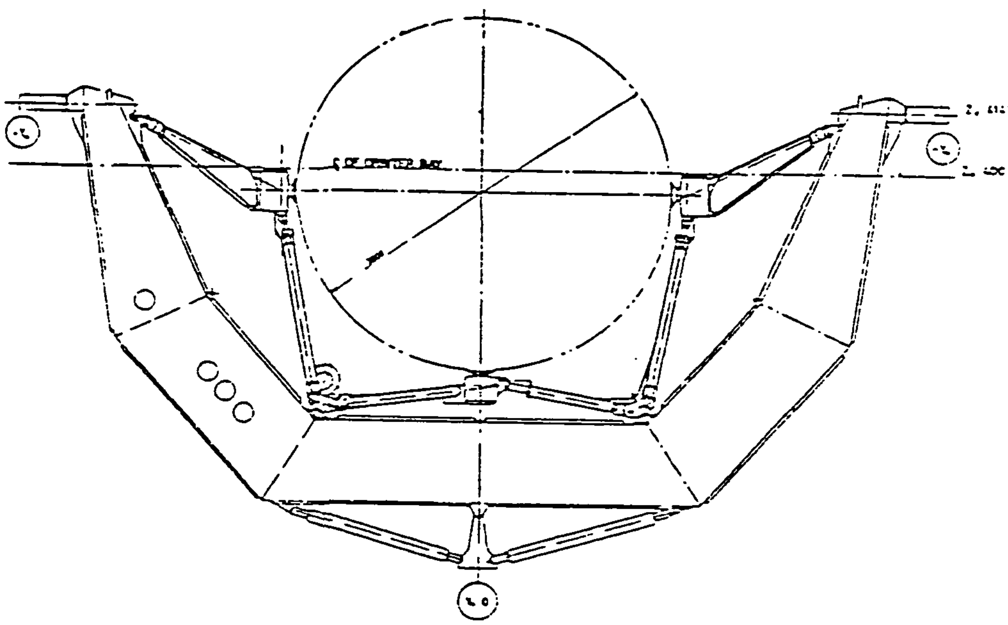

Figure 2.1.2.1.2-1: IPS PAYLOAD CLAMP ASSEMBLY

2.1.2.2 Thermal Concept

During orbital operations the IPS is capable of operating in the

- "cold case", that means in a completely shadowed configuration (worst case) for indefinite time
- "hot case", that means in a continuous full sun illumination (worst case) for a minimum time of 195 minutes when starting with the status after 9 hours cool-down. The time period for cooling off after maximum solar exposure - IPS stowed and units switched off - is 9 hours maximum. For not as extreme environmental conditions, the operational time of IPS is unlimited or can be enlarged. For hot cases TCA puts the following constraints on payload operation:
  - o The total solar and infrared radiation energy which is absorbed by the OSP radiator may not exceed 1.3 kWh during one hot operational phase. This could mean a roll angle restriction.
  - o During non-operation phases, payload and IPS shall remain attached. The maximum time with separated payload (stowed in PCU) shall not exceed 2 hours in sun phase (hot case).

For other not as extreme orbital conditions than the specified design cases of IPS TCA, the restrictions may be reduced or inapplicable depending upon the results of the mission dependent thermal analysis.

The TCA consists of the following components:

- Multilayer Insulation (common with Spacelab module)
- Radiators covered with second surface mirror foils
- White paint for not insulated external surfaces
- Black paint and tapes for trimming of radiative heat transfer
- Interface fillers for improvement of contact conductance
- Heater mats for heat leak compensation
- Thermistors for heater control and temperature monitoring
- TCA software loaded in CDMS for thermistor signal transformation, heater switching, temperature out-of-limit control and warning.

The 37 thermistors and the 29 heater loops are conditioned by the PEU and the DCU via the S/S-RAU and the IPS-RAU.
The heaters are switched on/off if the temperature is lower/higher than a pre-selected switching limit.

There are three different thermally relevant operation modes:

a) Operational Mode

The operational/in-calibration limits of all IPS units are automatically controlled by heater logic HL-A.

b) Stand-by Mode

The switch-on temperatures of not operated units are automatically controlled by heater logic HL-B (IPS stowed).

c) Non-Operating Mode

IPS is stowed and all units switched off. Every 30 minutes all temperatures have to be checked by HL-B. For temperatures colder than the switch-on limits the stand-by mode must be initiated. If max. operating limits are exceeded, the Orbiter must be turned to a cold attitude.

In order to achieve an optimized cool-down of IPS after a hot mission phase certain switch-off/on sequences of IPS units are to be performed which also provide operational conditions at the end of the cool down phase.

Ascent Constraints

- Pre-launch temperatures of IPS shall be not higher than 30°C when starting into a long duration worst hot ascent
- IPS main power shall be available at least 1 hour after opening of cargo bay doors
- After a worst hot ascent, the Orbiter shall not remain more than 1.5 hours in the worst hot attitude (Z solar inertial, full sun orbit) after cargo bay door opening.

Descent Constraints

The thermal conditions of IPS components which are to be realized by pre-descent orbital conditions before a descent is initiated, shall be evaluated by the mission dependant analysis. However as a minimum the IPS design enables a descent to be made with initial temperatures (when main power is switched off one hour before cargo bay door closure):

a) which are higher than those occuring after being 1.5 hours in the hot case transient (hot descent)

b) which are below those occuring after being 2 hours in the cool down period (cold descent).

An emergency descent can be initiated with steady state cold case and end of hot case transient temperatures without causing a failure which would lead to loss of personnel or damage of Spacelab or Orbiter.

If the temperatures of the pallets and the Orbiter radiators are in the range of $-20^{\circ}C$ to $+50^{\circ}C$ at cargo bay door closure, no temperature problems exist for IPS components.

2.1.2.3 Electrical Concept

The IPS electrical concept is determined by the extensive interface to the SL Command and Data Management Subsystem (CDMS) and to the Electrical Power Distribution Subsystem (EPDS). A blockdiagram of the IPS and its interfaces to SL is shown in Figure 2.1.2.3-1.

Clearly indicated are the main IPS electronic systems:

- the Power Electronic Unit (PEU) connected to the EPDS and
- the Data Control Unit (DCU) connected to the CDMS.

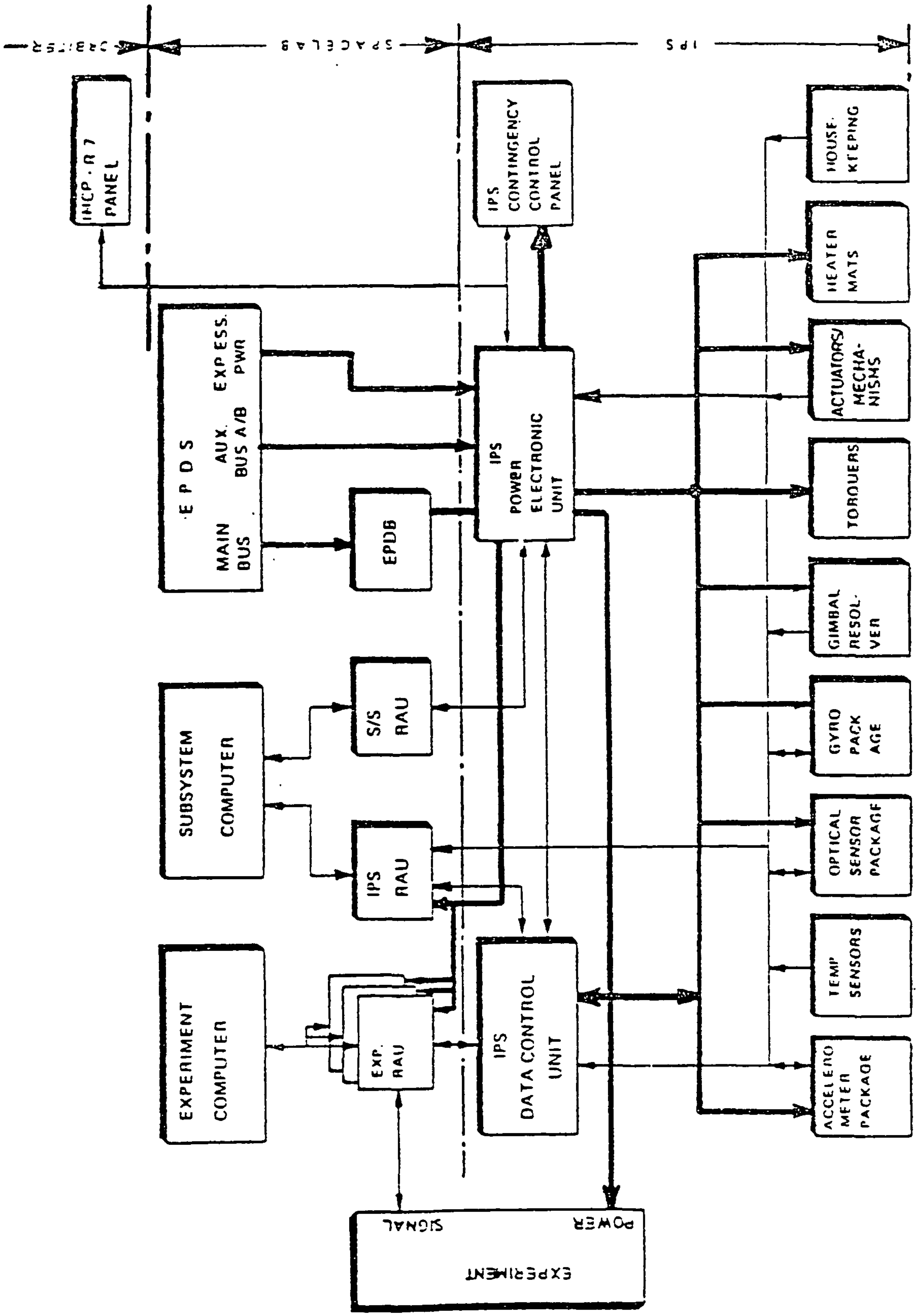

FIG. 2.1.2.3-1: IPS ELECTRICAL BLOCK DIAGRAM

2.1.2.3.1 IPS Power Electronic Unit

The IPS Power Electronic Unit (PEU) receives its primary power from the Spacelab Electrical Power and Distribution Subsystem (EPDS) by the following DC busses:

- Main DC Power from the Spacelab Electrical Power Distribution Box (EPDB) by three independent busses:
  - o DC1 Power Bus for IPS experiments
  - o DC2 Power Bus for IPS and RAU's
  - o DC3 Power Bus for heaters, RAU's and IPS OSP

- Auxiliary Power for contingency operation of IPS (retraction of IPS payload or activation of jettison):
  - o Auxiliary Power Bus A
  - o Auxiliary Power Bus B

- Experiment Essential Power

  from Spacelab Emergency Bos as a power source for experiments redundant to the experiment main power bus DC1.

The PEU provides the distribution and fusing of Spacelab power to the IPS electrical and electronic equipment, to the IPS or payload mounted Spacelab equipment (RAU's) and to the IPS payload.

Functionally the PEU is accomplishing the following tasks (see Electrical Blockdiagram, Fig. 2.1.2.3-1):

Unregulated DC main or essential power will be supplied to IPS experiments, unregulated main power to RAU's, Optical Sensor Package (OSP), Gyro Package (GP), IPS Data Control Unit (DCU).

Regulated DC power will be supplied to Accelerometer Package (ACP) and Jettison.

DC power at limited voltage and current at both polarities will be supplied to the actuator motors.

Controlled power will be supplied to torque motors and heater mats.

In the nominal operation mode the PEU receives control signals from the DCU to operate the torque motors.

For contingency operation of IPS during loss of main power or during nonoperation of CDMS the retraction of the IPS payload will be initiated by the IPS Contingency Control Panel (CCP) which is mounted in the Orbiter Aft Flight Deck. In this case the retraction circuitry for the torque motors is powered by the Spacelab Auxiliary Bus and controlled by the PEU internal "stowage loop".

In case that a safe landing of the Orbiter may be prevented by any failure of IPS, its separation from the orbiter is feasible from a separate section of the CCP, after the jettison function is enabled by a switch on the IMCP-R7 panel in the Orbiter Aft Flight Deck.

During deployment or retraction the PEU, controlled by the CDMS or the IPS CCP is driving the IPS mechanisms PCM, GLM and PGSM.

Furthermore the PEU is supporting the thermal control by heater mat switching initiated from the CDMS or by conditioning of the thermistor signals.

The power data given in Table 2.1.2.3-2 apply for the different power buses available at the S/L interface.

Spacelab Power Bus	IPS Power Consumer	Voltage Range [Volts]	Power Available at S/L Interface Mean [Watt]	Peak [Watt]
Main DC Power				
- DC1 Power Bus	IPS Experiments	24,0 - 32 (1) 23,5 - 32 (2)	-	1440 (1)
- DC2 Power Bus	IPS Electronics IPS Heater IPS RAU		- 500	1410 (2) 1400
- DC3 Power Bus	EXP RAU's		350	1200
Experiment Essential Power	IPS Experiment	21,5 - 32	-	100
Auxiliary Power	Jettison & IPS Set 1 Electronics	24,7 - 32	-	(3)

Note:

(1) IPS on pallet 1 through 3

(2) IPS on pallet 4 and 5

(3) 8 AMPS MAX continuous, after Spacelab and payload have been configured to the low power mode

Table 2.1.2.3-2: IPS Power Budget

2.1.2.3.2 IPS Data Control Unit

The IPS interface to the Spacelab CDMS is provided by the IPS Data Control Unit (DCU). The DCU is interfacing with the Spacelab Subsystem Computer (SSC) via the IPS RAU and corresponds with the Spacelab Experiment Computer (EXC) via the payload mounted EXP RAU1. This DCU serial data link provides the data exchange capability between the IPS experiments and IPS. The DCU controls the IPS data and command flow and processes the fast loop portion of the IPS pointing control loop by means of a minicomputer.

Controlled by the CDMS the DCU provides thermal control of IPS.

2.1.2.4 Software Concept

IPS software utilizes two processing units:

- the Spacelab Subsystem Computer (SSC) and
- the DCU "mini-processor"

The overall control of the IPS is exercised from the SL Data Display System (DDS) and subsystem computer via IPS application software except for the emergency stowage and jettison functions. The latter functions are commanded from the IPS contingency control panel (CCP) in the Orbiter Aft Flight Deck (AFD) and require no software.

Stability control of the payload in all 3 axes is processed within the IPS mini-processor "Data Control Unit" (DCU), based on error signals of rate integrating gyros (feedback-control) and an Accelerometer Package (feedforward-control).

The processing of pointing commands is performed in the SSC and the resulting desired attitudes and rates are inputs to the DCU.

Drift and attitude correction by means of real star/sun measurements with the aid of the optical sensor package containing 3 fixed head star/sun trackers is processed in the subsystem computer of the SL CDMS.

Furthermore it is possible to accept either attitude offset commands or replace the function of the boresighted star/sun tracker by an experiment sensor error signal via the SL experiment computer to the IPS DCU. However, the control (crew I/F) of these EXC functions is performed via the SSC.

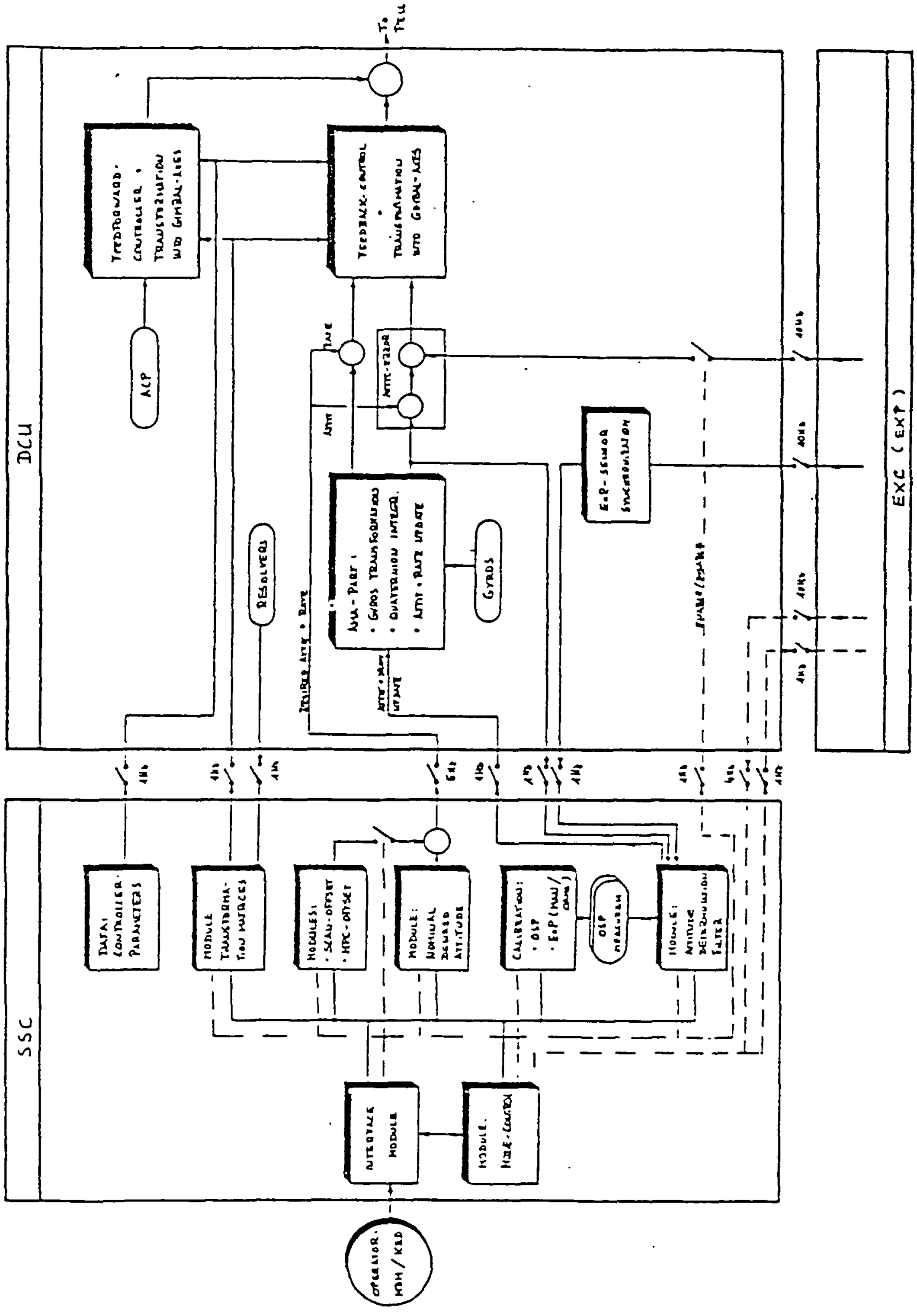

FIG. 2.1.2.4-1: IPS CONTROL LOOP BLOCK DIAGRAM

All these input data are processed in the DCU to give three control signals (one for each gimbal axis) and are sent to the Power Electronic Unit (PEU) which drives the DC torquers in the three gimbal axes.

The partitioning between SSC and DCU software is shown in the blockdiagram of Fig. 2.1.2.4-1.

The IPS related SW can be divided into two major components:

- Fast Loop SW

  residing in the DCU and performing the basic calculation of the inner control loop of the IPS for inertial pointing.

- Slow Loop SW

  residing mainly in the SSC except for the experiment-generated sensor data (replacing OSP BS data) and offset commands, which are input from the Experiment Computer (EXC).

The Slow Loop SW generates all mode option dependent desired and/or corrective attitude data as input to the Fast Loop SW in the DCU.

Fig. 2.1.2.4-2 shows an overview of the IPS SW environment and the data interfaces.

In the scope of this paper the right branch of Fig. 2.1.2.4-2, i.e., the SSC SW-DCU control loop is of main interest.

The EXC slow loop interface with the DCU is controlled via the SSC by enabling the EXC control cmd inputs in the DCU. For the operational functions which employ this interface (EXP CTL and

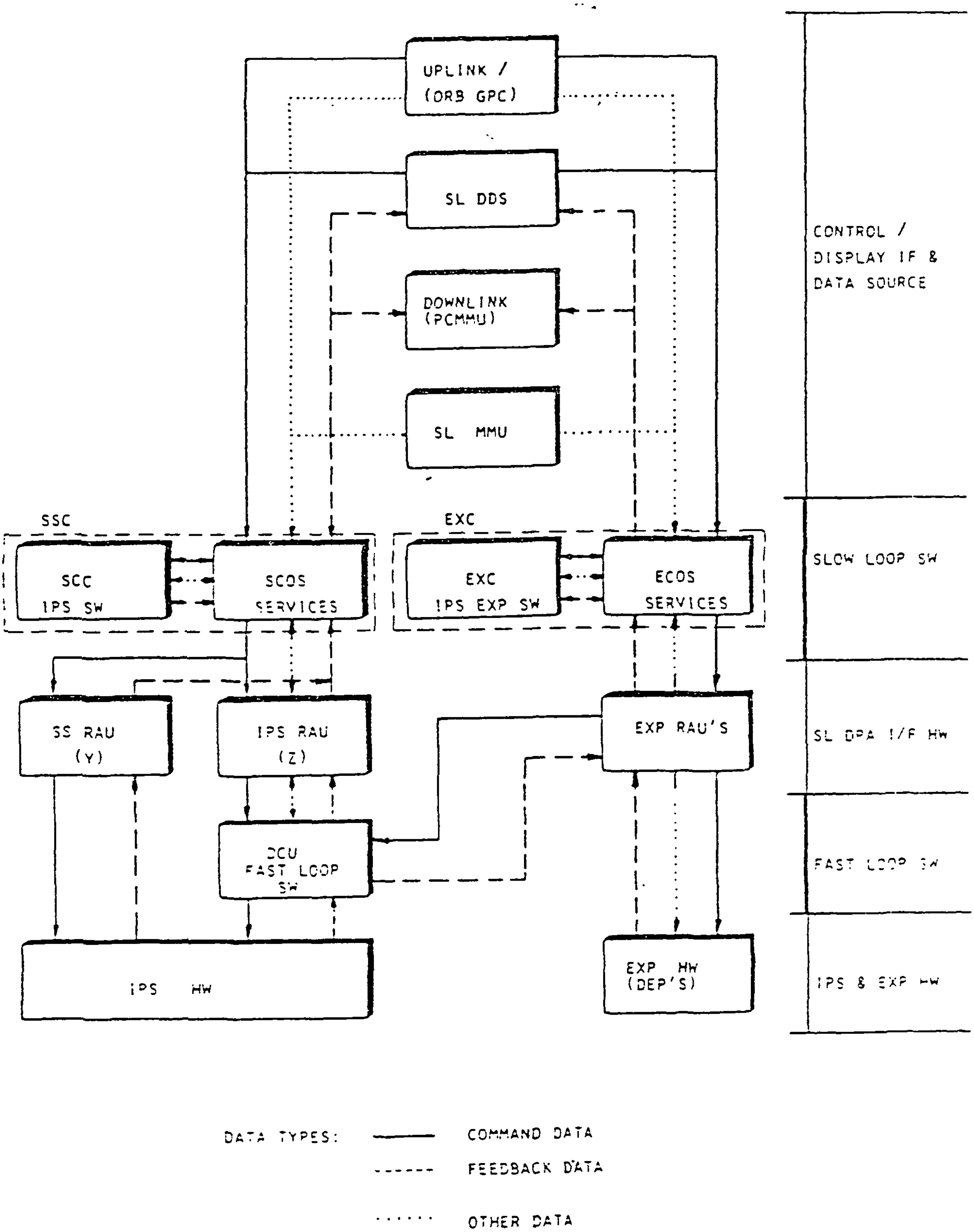

FIG. 2.1.2.4-2: IPS SOFTWARE ENVIRONMENT/DATA FLOW

EXP CMD) it is assumed, that by a TBD experiment/EXC operation the interface is configured correctly and needs only to be enabled/disabled via the SSC slow loop. Output data from the DCU to the EXC are partially enabled by activation of the DCU SW.

DCU Software

The DCU SW is subdivided mainly into two SW packages:

- the DCU Fast Loop SW
- the DCU Test SW:

The Fast Loop and Test SW run within the DCU data processor loaded and started by IPS application SW. Only one program is in the DCU memory at one time.

DCU Fast Loop SW

The main task of the DCU is to perform the calculation of the inner control loop of the IPS, done by the fast loop program. This program provides the capability of the IPS to keep the inertial attitude based on fast sensor information delivered from gyros (100 Hz) and accelerometers (50 Hz). Additionally it acquires attitude and drift signals based on 1 Hz optical sensor information processed within the SSC by the IPS Application SW. The DCU then updates every 40 ms its output to the PEU, which drives the gimbal torquers. The data between the DCU and the CDMS (i.e. SSC and EXC) are interfaced via the serial RAU data link.

DCU Test SW

The DCU Test SW executes a selftest of the DCU data processor.

SSC IPS Application Software (IPS SW)

The IPS flight SW consists of one functional configuration (FC).

Within the flight FC, the IPS SW comprises different memory configurations (MCs):

- 1 MC for ACT/DEACT mode,
- 8 MCs for stellar mode,
- 5 MCs for solar mode,
- 1 MC for earth mode.

The structure of all IPS flight MCs is quite similar. Each MC for flight application contains:

- the always core resident task for 10 Hz communication (PCOMM-task) between SSC and DCU
- the always core resident task for temperature control (TCA), MMU-access and GIMBAL HOLD (PTGGI-task),
- one MODE-task controlling different operational options (PMXXX-task),
- one KBD/ITEM-task supporting the operators/IPS SW interface (PINXX-task),
- one OSP-dialogue task (POXX-task),

- one AMA-task (PAXXX) which supports the attitude determination (Star Identification, Attitude Determination Filter)
- one SCHEDULE task (PSCXX-task) required to allocate the Interrupt Levels and to start all tasks of a MC.

To allow for required operations, each task comprises up to 6 internal programs, each supporting a special function.

Thus each MC supports a dedicated operational function identifiable from the respective acronym as follows:

MC Name	Operational Function/Mode
ACDEAC	Activation/DEActivation mode
SLEWST	SLEWing in STellar mode
IDINST	Star IDentification/INitial in STellar mode
OSPCST	OSP Calibration in STellar mode
IDOPST	Star IDentification/OPerational in STellar mode
OHSCST	Optical Hold plus SCan in STellar mode
OHMPST	Optical Hold plus MPc in STellar mode
SCMPST	SCan plus MPc in STellar mode
OHOAST	Optical Hold plus OAms in STellar mode
SLEWSO	SLEWing in SOlar mode
IDINSO	star sun IDentification/INitial in SOlar mode
OHSCSO	Optical Hold plus SCan in SOlar mode
OHMPSO	Optical Hold plus MPc in SOlar mode
SCMPSO	SCan plus MPc in SOlar mode
SCMPEA	SCan plus MPc in EArth mode

Display Processing Concept

All data necessary to operate IPS during flight are provided on fixed format displays (FFDs).

The IPS FFDs basically utilize the SCOS display services.

The respective FFD is only operative as long as the associated MC is loaded.

Command Processing Concept

IPS utilizes the SCOS capability for keyboard inputs and related SCOS command processing to issue individual commands for IPS operations.

By uplink commands to the IPS via the Orbiter (MDM uplink), the ground has the same functional capability to control IPS that the crew has in using the keyboard.

The IPS SW is designed such that "parallel" operator inputs are accepted by SW. Every task which has to receive commands for IPS applications SW sets a request in the first run after having been started. Therefore, the IPS application SW accepts any command delivered by SCOS from the KBD (Item-, FK entry), the ground via MDM or the EXC via the DCU for execution at any point in time on the basis of a 1 Hz repetition cycle with the assumption that no 2 command inputs from the same source occur less than 1 second apart.

2.1.2.5 Operations Concept

Operational Mode Diagram

The IPS operational mode diagram in Fig. 2.1.2.5-1 gives a schematical overview, how the various IPS operations are functionally processed.

Two generic processing blocks

- Desired Attitude Processing
- AMA Processing (Attitude Measurement Assembly)

provide the inputs to the IPS dynamics which allow to perform the required pointing modes with the required accuracy.

The desired attitude processing allows to select, acquire and hold desired attitude (including offsets), which are defined in stellar, solar or earth coordinate systems or relative to the Orbiter.

Without AMA processing, the desired attitude acquisition and -hold is based on IPS gyro data only (with static, stored drift updates).

The AMA processing serves to increase the pointing accuracy by experiment dependent static calibration and/or dynamic attitude updates based on attitude data from the OSP or from experiment sensors (including OAMS).

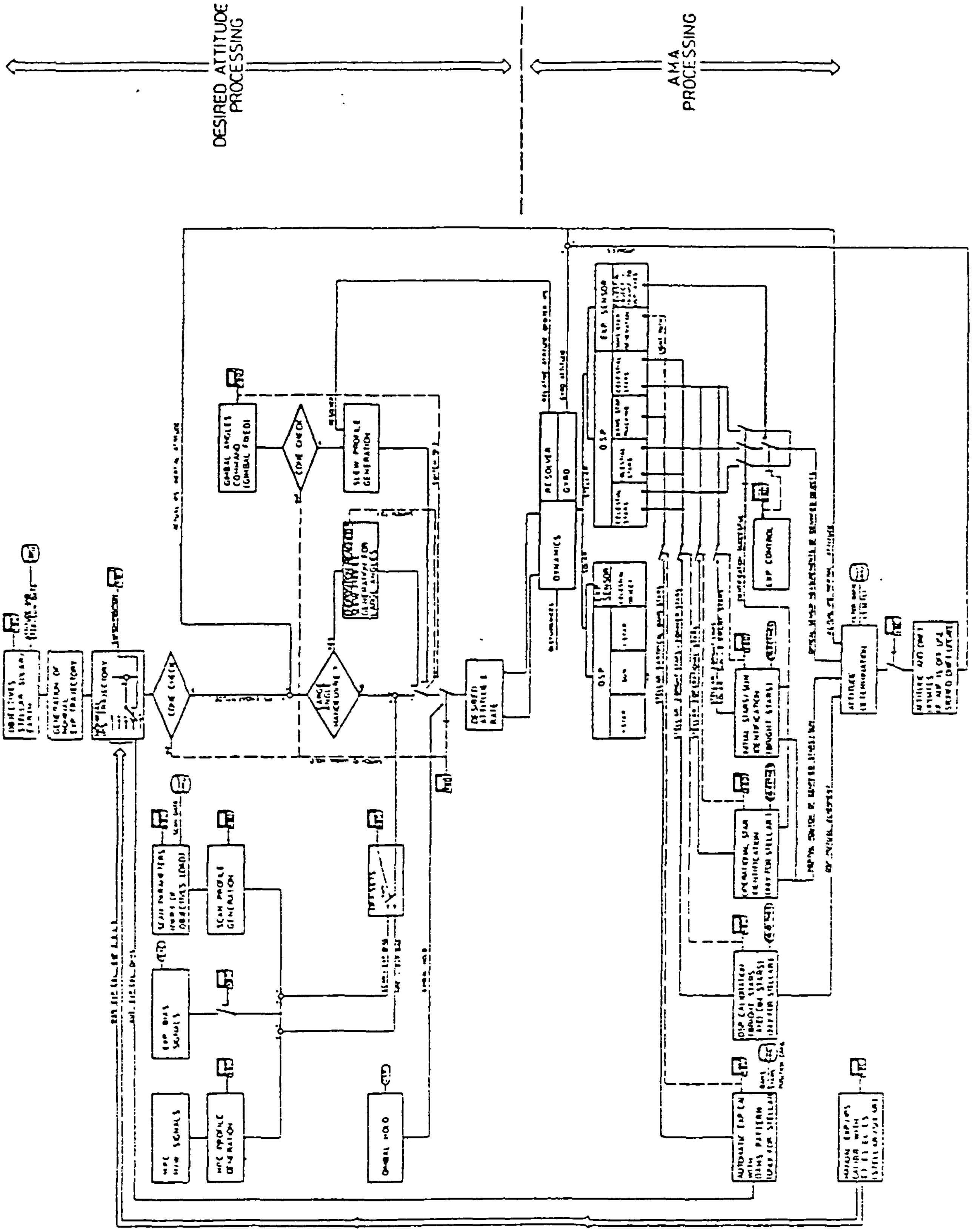

FIG. 2.1.2.5-1: OPERATIONAL MODE DIAGRAM

Groundrules for IPS On-Orbit Operations

- IPS will be inactive during the ascent/descent
    - o No normal or contingency IPS operations will be performed during these phases

- The IPS on-orbit activities are performed in the on-orbit period between end of SL activation and start of SL deactivation.

- During this phase IPS is in one of the three states:
    - o inactive
    - o operated in a normal mode
    - o operated in a contingency mode

  All flight operations to achieve and maintain the three states will be covered by related IPS sequences.

- All normal IPS operations are performed from an SL DDS, employing ITEM-commands and feedbacks on dedicated IPS FFDs in the SSC. Unlike the basic Spacelab, which is primarily operated via the Orbiter GPC CRT and Keyboard, the GPC is not involved in IPS operations.

- Although the ground (MCC/POCC) has the same command capability as the crew has on board, the ground control of IPS will be procedurally restricted to objective loads and modifications as well as start and stop of experiment dedicated IPS operations (scan, exp. command and exp. control) for solar and stellar missions. Activation/deactivation as well as earth pointing will always be performed by the crew.

IPS Operational Functions

Definition:

An operational function is an IPS operation or operational mode, which - success oriented - is always performed or established and maintained as an operational entity.

In general, an operational function is initiated and terminated procedurally by the IPS operator.

The operational functions are grouped into the five categories:

- Set-up
- Attitude Control
- Acquisition
- Offset Pointing
- Experiment Support

Table 2.1.2.5-1 identifies all IPS operational functions, the assignment to the functional categories and their applicability in the basic modes.

F1: Activation

During Activation the Orbiter- and SL subsystem resources are provided, IPS equipment is activated, the IPS and its payload are attached, unclamped and slewed into the upright position.

F2: Deactivation

During deactivation the IPS is slewed into the stowed position, IPS and its payload are separated and clamped and all IPS equipment is deactivated.

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Operational Functions Overview			Basic IPS Mode				Funct. #
Category	Operational Function	Short Name	ACT/ DEACT	Stellar	Solar	Earth	
Setup	Activation	Activation					F1
	Deactivation	Deactivation					F2
	IPS Standby	IPS STBY					F18
	DCU Test & Dump	DCUT					F19
Attitude Control	Objective Load and/or Modification	OBJ LOAD/MOD					F5
	Attitude Command	ATT CMD					F6
	Gimbal Angle Command	GMBL ANG CMD					F4
	Gimbal Hold (on Orb. State Vector)	GMBL HOLD					F3
Acqui-sition	Initial Stellar Identification	ST INIT ID					F7
	OSP Calibration	ST OSP CAL					F20
	Initial Solar Identification	SO INIT ID					F8
	Operational Star Identification	ST OPNL ID					F9
	Attitude Hold (Gyros Only)	ATT HOLD					F12
	Optical Hold (Gyros + OSP)	OPT HOLD					F13
Offset Pointing	Scan	SCAN					F15
	Manual Pointing Control	MPC					F16
Experiment Support	Experiment Bias Command	EXP CMD					F17
	Experiment Control	EXP CTL					F14
	Manual Experiment Calibration	MAN EXP CAL					F10
	Experiment Calibration With OAMS	EXP CAL OAMS					F11

TABLE 2.1.2.5-1: IPS OPERATIONAL FUNCTIONS

F3: Gimbal Hold (GMBL Hold)

GMBL Hold commands IPS to hold the current gimbal angles relative to the Orbiter. This is accomplished by use of the Orbiter state vector (SV), superimposing the rotation of the Orbiter on top of the IPS inertial hold.

GMBL Hold is primarily a contingency option, available throughout any IPS operation when IPS is unstowed. During normal IPS operations it is employed automatically during mode-transition phases to exclude uncontrolled motions of IPS.

F4: Gimbal Angle Command (GMBL ANG CMD)

GMBL ANG CMD allows to command IPS gimbal angles relative to the Orbiter. It can be performed in any basic mode, it is always performed as large angle manoeuvre, i.e., the slew SW must be loaded, upon GMBL ANG CMD IPS acquires the commanded gimbal angles and holds these based on resolver data. In Stellar, Solar and Earth mode a plausibility check is performed which sets IPS into GMBL Hold if the cone limits would be violated. In the ACDEAC mode this check is omitted to allow stowage/erection.

F5: Objective Load and/or Modification (OBJ LOAD/MOD)

OBJ LOAD/MOD is an essential precondition for any IPS Stellar, Solar or Earth pointing. Objective data including optional scan data are loaded from MMU, modified or restored from SSC core as desired objective.

The performance is functionally similar for all three basic pointing modes.

F6: Attitude Command (ATT CMD)

ATT CMD acquires a desired objective which is defined by a preceeding OBJ LOAD/MOD operation in the Stellar, Solar, Earth (Local Vertical) or Earth (Landmark) coordinate system. During execution of ATT CMD, IPS will be moved until the current attitude equals the desired.

F7: Initial Stellar Identification (ST INIT ID)

After launch or long periods of attitude hold on Gyros only, ST INIT ID has to be performed to run the strapdown attitude determination system which realignes the inertial IPS attitude. INIT ID identifies unique bright stars only.

F8: Initial Solar Identification (SO INIT ID)

SO INIT ID identifies a solar target (sun + bright start). After the identification process the strapdown attitude determination is performed which realignes the IPS inertial attitude.

F9: Operational Star Identification (ST OPNL ID)

The operational star identification is performed in stellar mode only using a set of operational stars loaded as part of an objective load.

During the operational star identification process no attitude updates are performed i.e. IPS is in attitude hold on gyros only.

F10: Manual Experiment Calibration (MAN EXP CAL)

MAN EXP CAL allows to determine the actual alignment between the EXP LOS and the IPS LOS. Employing attitude corrections until an EXP-provided output indicates the optimum attitude, the predetermined alignment matrix (or unity matrix if not predetermined) is updated with the optimum values and will statically be used when IPS is pointing in support of the respective experiment.

F11: Experiment Calibration with OAMS (EXP CAL OAMS)

IPS SW determines and updates the alignment of one experiment versus the boresighted sensor during stellar mode operation using the experiment provided On-orbit Alignment Measurement System (OAMS). In this option the boresighted sensor continuously tracks five artifical OAMS stars. The difference between the expected (ground determined) and the actual location of the artificial stars in the BS tracker field of view is used to update the specific experiment alignment matrix dynamically.

F12: Attitude Hold (ATT HOLD)

In ATT HOLD IPS points fixed relatively to the basic coordinate system as listed below:

- in stellar mode: inertial mission true of date,
- in solar mode: solar ecliptic reference system,
- in earth mode (Local Vertical): local vertical reference system
- in earth mode (Landmark): geodetic reference system

ATT HOLD is based on Gyro data only. If provided, the last update of the drift estimation will be used statically for drift compensation.

F13: Optical Hold (Gyros plus OSP) (OPT HOLD)

High pointing accuracy and stability control are not only based on gyros only, but additionally on optical sensor package (OSP) updates which provide for a gyro drift compensation by means of an absolute attitude reference signal. Therefore, the optical hold is an attitude hold under gyro control with optical sensor update to the strapdown attitude determination system, invoked automatically or by operator command after acquisition and identification of a celestial target, OPT HOLD is provided only in stellar or solar mode.

F14: Experiment Control (EXP CTL)

IPS accepts two axis control commands derived from the experiment sensor measurements in stellar or solar mode. The experiment provided sensor replaces the measurements of the OSP boresighted sensor. Experiment data are transmitted to IPS via the Experiment Computer interface. Under experiment control IPS uses for the strapdown-filter a prelaunch determined set of filter constants, defined for each experiment.

To achieve the full IPS bias-pointing-performance capability, the experiment sensor must track the same object (star/sun) as the IPS boresighted sensor.

F15: Scan

In all three basic pointing modes IPS is capable to superimpose a scan on top of attitude - or optical hold (plus optional EXP CMD).

When scan is started, IPS scans a rectangular field according to scan parameters defined in the desired objective data.

The operator has the option to start, stop, interrupt/resume the scan or define the current position as new scan center.

F16: Manual Pointing Control (MPC)

In all three basic pointing modes IPS is capable to superimpose MPC manoeuvres on top of attitude or optical hold (plus optional EXP CMD). MPC allows the crew to manually command yaw, pitch and roll signals. Rate commands generated by the handcontroller are superimposed on rate commands from other enabled options. Inertial attitude commands are derived from the total rate command.

The crew can change the maximum MPC rate for yaw and pitch which also defines the max. roll rate via a premission defined scale factor, furthermore, the crew can select medium and low rates.

F17: Experiment Bias Command (EXP CMD)

In all three basic pointing modes IPS is capable to superimpose three axes experiment bias commands on top of attitude or optical hold (plus optional scan or MPC). These experiment data are transmitted to the DCU via the experiment computer in-

terface. Experiment off-set commands are only accepted by the IPS SW when this option has been selected by a separate keyboard command from the SSC.

F20: Stellar OSP Calibration (ST OSP CAL)

By tracking in each tracker of the OSP two identified stars (1 bright and 1 dimmer star) simultaneously, the OSP calibration is performed. This compensates the alignment errors of the OSP skewed versus the boresighted sensor.

2.1.2.6 Safety Concept

2.1.2.6.1 General

Safety of human life has the highest priority during all operational phases of IPS. In particular, special emphasis is given to crew safety during ascent/descent and orbital operations. Therefore catastrophic events must be excluded under all circumstances to prevent:

- loss of personnel and/or
- loss of Orbiter or Spacelab.

From this the following top catastrophic hazards exist a priori for IPS:

- hitting the Orbiter or Spacelab
- inability to configure for safe return.

In consequence the IPS design has payed special attention to the safety critical functions:

- prevent hitting the Orbiter or Spacelab
- configure for safe return.

Before the occurance of an event, the hazard (i.e. potential of occurance of a hazardous situation) is latent and exists regardless of the introduction of means to reduce the probability of the event. The level of an hazard cannot be changed, the hazard can only be controlled by introducing appropriate means (safeguards).

The top catastrophic hazards and the corresponding safety critical functions for IPS are shown in the following table. To control the top catastrophic hazards, means must be available to preclude the occurance of the hazardous events. Such safeguards are shown in the table. Causes of the top catastrophic hazards are identified and the relevant controls are defined.

top catastrophic hazard	safety critical function	safeguards
A. hitting the Orbiter/SL	prevent hitting Orbiter/SL	- bumper device
B. inability to configure for safe return	configure for safe return	- stowage via CDMS - stowage via CCP - jettison - EVA
C. premature jettison	prevent premature jettison	- appropriate inhibits o RPC switch o ARM function o EXECUTE function

Control of Hazard A

For hazard A the introduction of the passive, mechanical bumper device provides adequate control for the occurance of the hazard as long as the PL is attached to IPS and provided the bumper is properly designed. In case of inadvertent operation of PGSM during pointing mode or in case of inadvertent operation of Payload Clamp Mechanism (PCM) or Gimbal Latch Mechanism (GLM) during ascent/descent the bumper device is no longer a safeguard against collision between IPS/PL and Orbiter/SL.

Control of Hazard B

In the case of hazard B a combination of various active means with a reliability less than 1 are used to control the hazard.

Besides the reliability a further argument for the introduction of several safeguards is the fact that not every safeguard is valid for all operational phases.

The standard operation to configure IPS/PL for safe return is stowage by using the Gimbal Latch Mechanism, Payload Gimbal Separation Mechanism and Payload Clamp Mechanism. For safety reasons there are 3 options to stow (Fig. 2.1.2.6.1-1):

- normal stowage (SET1 via CDMS)
- back-up stowage (SET2 via CDMS)
- contingency stowage (SET1 via CCP).

Control of Hazard C

The jettison device is introduced as a safeguard for hazard B. It generates itself the top catastrophic hazard C, named "premature jettison". To preclude the hazardous event certain design features (inhibits) are introduced, which are:

- RPC 2 switch
- ARM function
- EXECUTE funtion

Items introduced in order to close a top catastrophic hazard are per definition safety critical items. The identification of critical items will be based also on these top catastrophic events (hazards), i.e. items being involved in (or part of) functions which may result in top catastrophic events shall be considered primarily as a critical item and will be examined for its criticality.

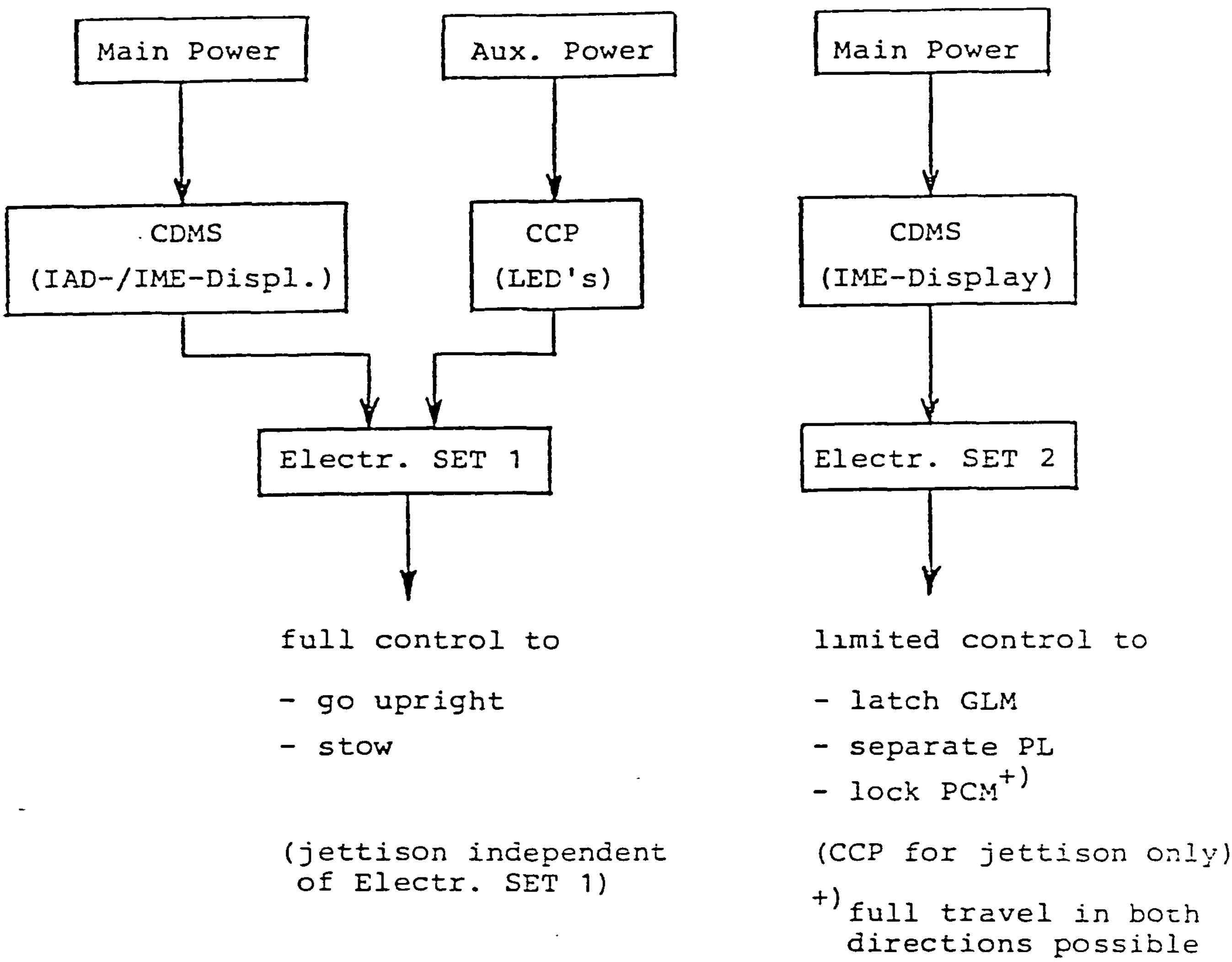


Fig. 2.1.2.6.1-1: OPTIONS FOR IPS ACTIVATION/DE-ACTIVATION

2.1.2.6.2 Safeguards for Top Catastrophic Hazards

For the redesigned IPS there is a significant safety improvement with regard to the top catastrophic hazard A: "hitting the Orbiter/SL" because of the

- addition of a bumper device which
  - o represents a passive mechanical stop for motions of IPS and its payload as long as the PL is attached to the IPS and the cargo bay doors are open
  - o is more reliable than the concept of the former design for active electronical limitation of range and rate (a failure of a LBP then resulted in an increased probability of collision with Orbiter/SPACELAB)
- elimination of two complex safety critical systems
  - o hardware range/rate limitation electronics
  - o LBP (brakes)
- elimination of items from the critical items list. By introduction of the bumper all items taking part in manoeuvres required to support the mission objective (i.e. manoeuvres) only, are no longer safety critical items. This does however not apply to those items which are required for configuration for safe return.

  It must be recognized that the bumper device itself is a safety critical item.

The safeguards for the top catastrophic hazard B: "inability to configure for safe return" are in the order of preference:

- back-up stowage (SET2) via CDMS
- stowage via CCP (SET1)
- jettison
- EVA

In the following these safeguards for the redesigned IPS are discussed:

- The principal features of the revised stowage concept which constitutes the primary function for hazard B are listed below:

  - o introduction of gimbal latch mechanism (GLM)
  - o elimination of load by-passes and formlocks within drive units
  - o rate limitations no longer necessary for safety reasons
  - o GLM less complex (more reliable) than brake design
  - o GLM redundant for the locking function (SET2)
  - o PGSM redundant for 'separate' (SET2)
  - o IPS during stowage in pointing mode (normal stowage) until GLM is locked
  - o redundant switches indicating locked position of EPF and GLM.

- The main features of stowage via CCP are:
  - o zero positioning axis by axis with removal of the control torque after zero position is reached in the relevant axis
  - o no locking of axes in zero position for roll and X-EL since effects introduced by disturbances (man motion, VCS firing and mass unbalanced about axes) are negligible compared to friction of bearings and CFT
  - o CCP stowage represents a back-up for normal stowage and stowage using SET2 via CDMS.

  Apart from those features no further modifications resulting from safety considerations have been introduced within the new design for contingency stowage.

- The main features of the jettison concept are:
  - o separation plane is located below the GLM.
  - o a third inhibit is introduced (RPC 2 switch on R7 panel).

  Since the LBP brakes are eliminated in the new design, a failure of the torquers may cause IPS to swing (supported by deflections at the bumper ring) for max. time of 20 min. until it stops. This constitutes a time constraint for the use of the jettison capability.

  No further modification have been introduced resulting from safety considerations into the jettison design.

- For EVA some modifications resulting from safety considerations compared to the baseline are introduced:
  - o deletion of the Drive Unit's EVA handles
  - o redesign of the EVA features of the PCA
  - o introduction of EVA Payload retention.

The control of the top catastrophic hazard "premature jettison" is provided by verification of JSC 08060 B requirements (Appendix I to Safety SR-IS-0002) and additionally by

- introduction of 3 inhibits
  (RPC 2 switch, ARM function, EXECUTE function).

2.1.3 Interfaces

2.1.3.1 Mechanical Interface

2.1.3.1.1 Interface with Orbiter

2.1.3.1.1.1 IPS Equipment located in the Orbiter Payload Bay

There is no direct mechanical interface between the Orbiter and IPS equipment located in the Orbiter Payload Bay. However, the performance of the IPS is influenced by the behaviour of the Orbiter. The Orbiter behavioural characteristics and dynamic model used to investigate the IPS performance and structural integrity are those defined in section 2.1.4.

2.1.3.1.1.2 IPS Equipment located in the Orbiter Aft Flight Deck

The only piece of IPS equipment located in the Orbiter Aft Flight Deck is the Contingency Control Panel.

2.1.3.1.1.3 Interfaces with Orbiter Tools

The interface between IPS and the Orbiter Tools for the manual opening of the jettison bolts is detailed in 20-ICD-IPS.

2.1.3.1.2 Interfaces with the Spacelab Pallet

IPS is attached to the pallet at the hardpoints and sill fittings. The IPS payload clamp assembly attachments are dependent on the payload characteristics. The IPS gimbal support structure (GSS) is attached to the pallet hardpoints, numbers 10, 12, 14 and 18.

2.1.3.2 Thermal Interface

2.1.3.2.1 Interfaces with the Orbiter

2.1.3.2.1.1 IPS Equipment located in the Orbiter Payload Bay

2.1.3.2.1.1.1 Thermal Design Configurations and Models

The Orbiter TMM used for the Spacelab Mission 2 thermal analyses is detailed in ES3-76-7.

Specular solar energy reflection from the forward Orbiter radiators is addressed in NASA TM-78270.

2.1.3.2.1.1.2 Mission Thermal Environment

The temperatures of the Orbiter and IPS elements used in and derived from the IPS baseline thermal analyses are shown in Table 2.1.3.2.1.1.2-1. These temperatures are those used in and derived from the IPS design cases defined in IF-IS-0001 and envelope the temperatures to be expected during the Spacelab Mission 2. The temperatures of the Orbiter and IPS elements resulting from the Spacelab Mission 2 thermal analyses are detailed in NASA-Ref-TBD3.

NODE No (4)	NODE DESCRIPTION \ TEMPERATURE (°C)	END OF ASCENT (1)		ON-ORBIT		DESCENT (3)	
		HOT	COLD	HOT (2)	COLD	HOT	COLD
1210	Gimbal Support Structure	44	- 10	99	-122	99	-26
1220	Gimbal Support Structure	44	- 4	76	-117	78	-21
1230	Gimbal Support Structure	44	- 4	76	-118	78	-21
1240	Gimbal Support Structure	45	- 4	75	-116	76	-21
1250	Gimbal Support Structure	45	- 4	76	-117	77	-21
2201	Framework/Gimbal Support Structure Connection	46	- 1	53	-114	54	-14
2202	Framework/Gimbal Support Structure Connection	45	- 1	59	-115	60	-15
2203	Framework/Gimbal Support Structure Connection	46	- 2	60	-114	61	-16
2204	Framework/Gimbal Support Structure Connection	45	- 1	60	-114	62	-16
2520	Framework/Elevation Drive Unit MLI	52	- 36	87	- 73	87	-18
2521	Framework/Elevation Drive Unit MLI	50	- 22	70	- 41	70	- 8
4511	XDU Cable Feedthrough MLI	45	- 10	66	-107	66	-24
4219	Yoke Radiator	44	0	53	- 9	55	10
4515	Yoke +X Sidewall MLI	47	- 27	90	-106	90	-32
6231	Equipment Platform SSM	57	- 47	43	- 60	64	-11
6232	Equipment Platform SSM	52	- 31	40	- 58	62	-16
6233	Equipment Platform SSM	45	- 12	36	- 62	59	-23
6236	Equipment Platform SSM	44	- 12	27	- 60	51	-24
6237	Equipment Platform SSM	52	- 31	31	- 59	55	-16
6238	Equipment Platform SSM	57	- 46	42	- 57	64	-11
6251	Payload Attachment Ring	49	- 35	47	-100	26	-17
6321	RAU/ICS Baseplate	46	1	29	- 3	49	- 1
6501	Equipment Platform MLI	54	- 55	109	-134	109	-32
6502	Equipment Platform MLI	54	- 55	109	-135	109	-33
6503	Equipment Platform MLI	53	- 49	110	-114	110	-29
6504	Equipment Platform MLI	53	- 49	110	-115	110	-29
6505	Equipment Platform MLI	47	- 18	108	-103	108	-26
6511	Equipment Platform MLI	56	- 69	47	-163	47	-34
6512	Equipment Platform MLI	42	- 20	49	-132	48	-45
9101	Payload	50	-120	70	-120	70	-20
9111	Pallet below outer longeron	40	- 20	93	-143	93	-62
9112	Pallet above outer longeron	40	- 20	93	-147	93	-62
9113	Sill	40	-148	85	-146	85	0
9121	Module Aft End Cone Top	60	-113	37	-164	37	-24
9122	Module Aft End Cone Bottom	48	- 74	61	-135	61	-30
9125	Module Top	65	-167	-54	-170	-53	-19
6320	RAU	46	1	30	- 4	50	- 1
6330	ICS	45	2	33	- 13	53	- 1
9131	Orbiter Aft Bulkhead Top	59	- 91	33	-152	34	-26
9132	Orbiter Aft Bulkhead Top	51	- 47	67	-158	67	-40
9141	Orbiter Radiators	66	- 20	15	- 20	15	-20
9142	Orbiter Wings	66	-194	55	-194		

Notes:
1) Defined as PBD open, SL Services available
2) Defined as end of 3.25 hrs operating in sun mode
3) Defined as end of descent
4) Node numbers from IPS thermal analyses

TABLE 2.1.3.2.1.1.2-1: ORBITER/SPACELAB/IPS INTERFACE TEMPERATURES

2.1.3.2.1.2 IPS Equipment located in the Orbiter Aft Flight Deck

The only item of IPS equipment mounted in the Orbiter Aft Flight Deck is the Contingency Control Panel (CCP). The CCP will be surface cooled by the cabin gas.

The dissipation of the CCP is 3 watts maximum. The temperature of the structural mounting interface and the mean radiant environment temperature is 49°C maximum.

2.1.3.2.2 Interfaces with the Spacelab

2.1.3.2.2.1 Thermal Design Configurations and Models

The Spacelab TMM used for the Spacelab Mission 2 thermal analyses is that shown in NASA-Ref-TBD5.

2.1.3.2.2.2 Structural Attachment Thermal Interfaces

The detail design of the mechanical connections between IPS and Spacelab is described in 2O-ICD-IPS.

2.1.3.2.2.3 Mission Thermal Environment

The temperatures of the Spacelab and IPS elements used are derived from the IPS baseline thermal analyses. These temperatures are those used in and derived from the IPS design cases defined in IF-IS-0001 and envelope the temperatures to be expected during the Spacelab Mission 2. The temperatures of the Spacelab elements and the corresponding temperatures of the IPS elements resulting from the Spacelab Mission 2 thermal analyses are detailed in NASA-Ref-TBD7. In particular the thermal environment for the Spacelab Remote Acquisition Unit and Interconnect stations mounted on the IPS is shown in Table 2.1.3.2.2.3-1.

	Mean Radiant Environmental Temperature	Mean Enrivonmental Emissivity
Hot Case	+ 60°C	0.9
Cold Case	- 40°C	0.9

Table 2.1.3.2.2.3-1: RAU and IS Thermal Enrivonment

2.1.3.3 Electrical Interface

The electrical interface between IPS and Spacelab is descripted in Fig.'s 2.1.3.3-1 to -4.

DORNIER

Dornier System GmbH

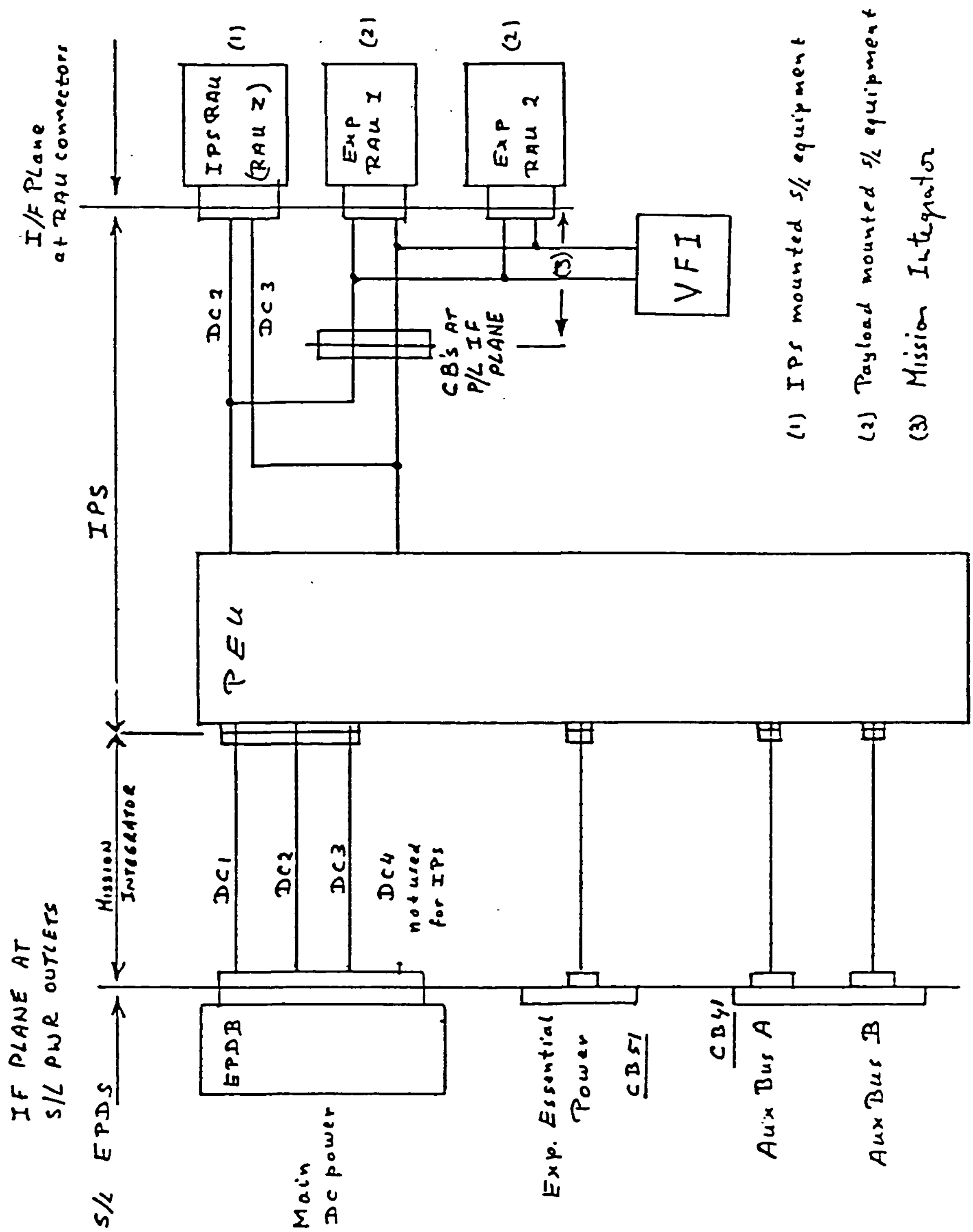


Fig. 2.1.3.3-1 IPS-SPACELAB POWER INTERFACES

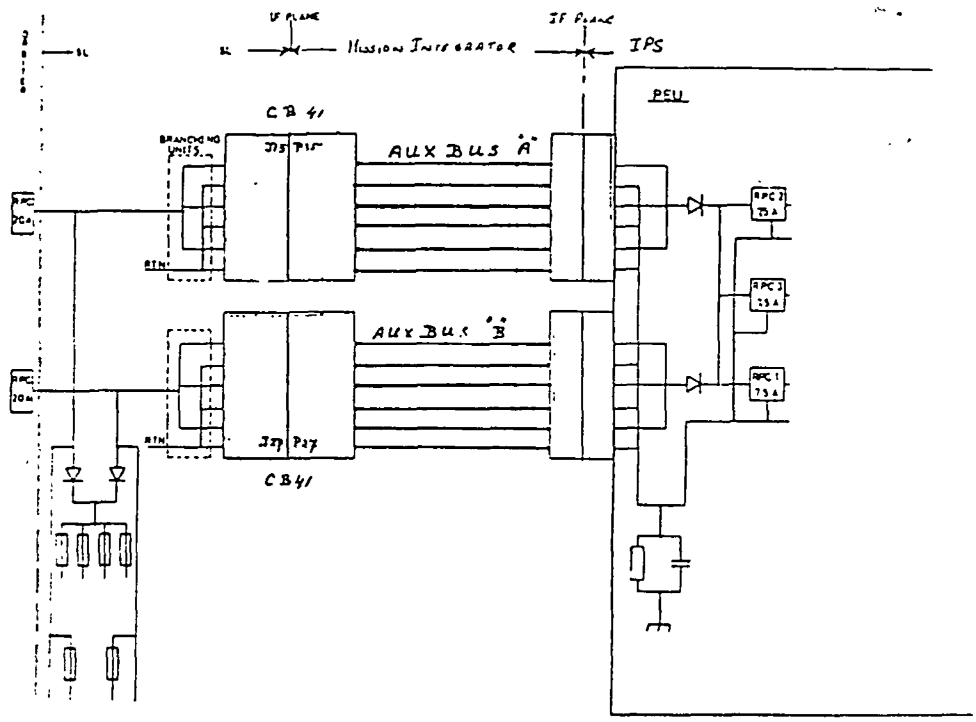


FIG. 2.1.3.3-2: AUXILIARY POWER INTERFACE

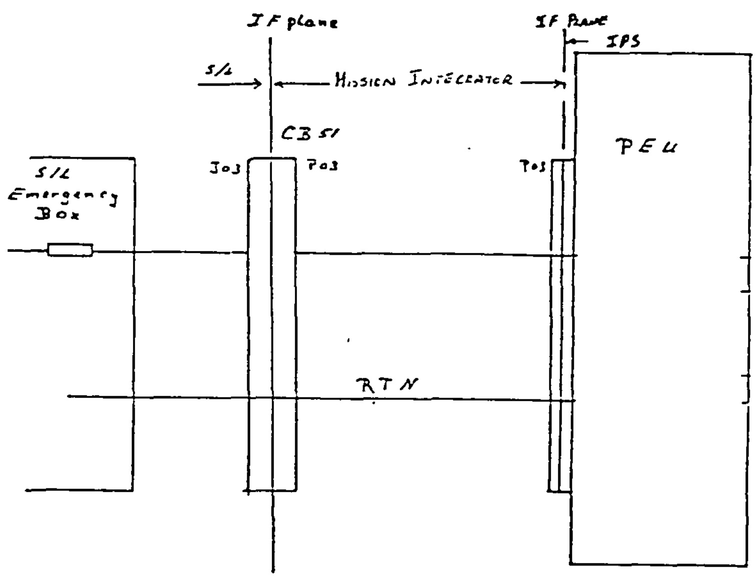


FIG. 2.1.3.3-3: EXPERIMENT ESSENTIAL POWER INTERFACE

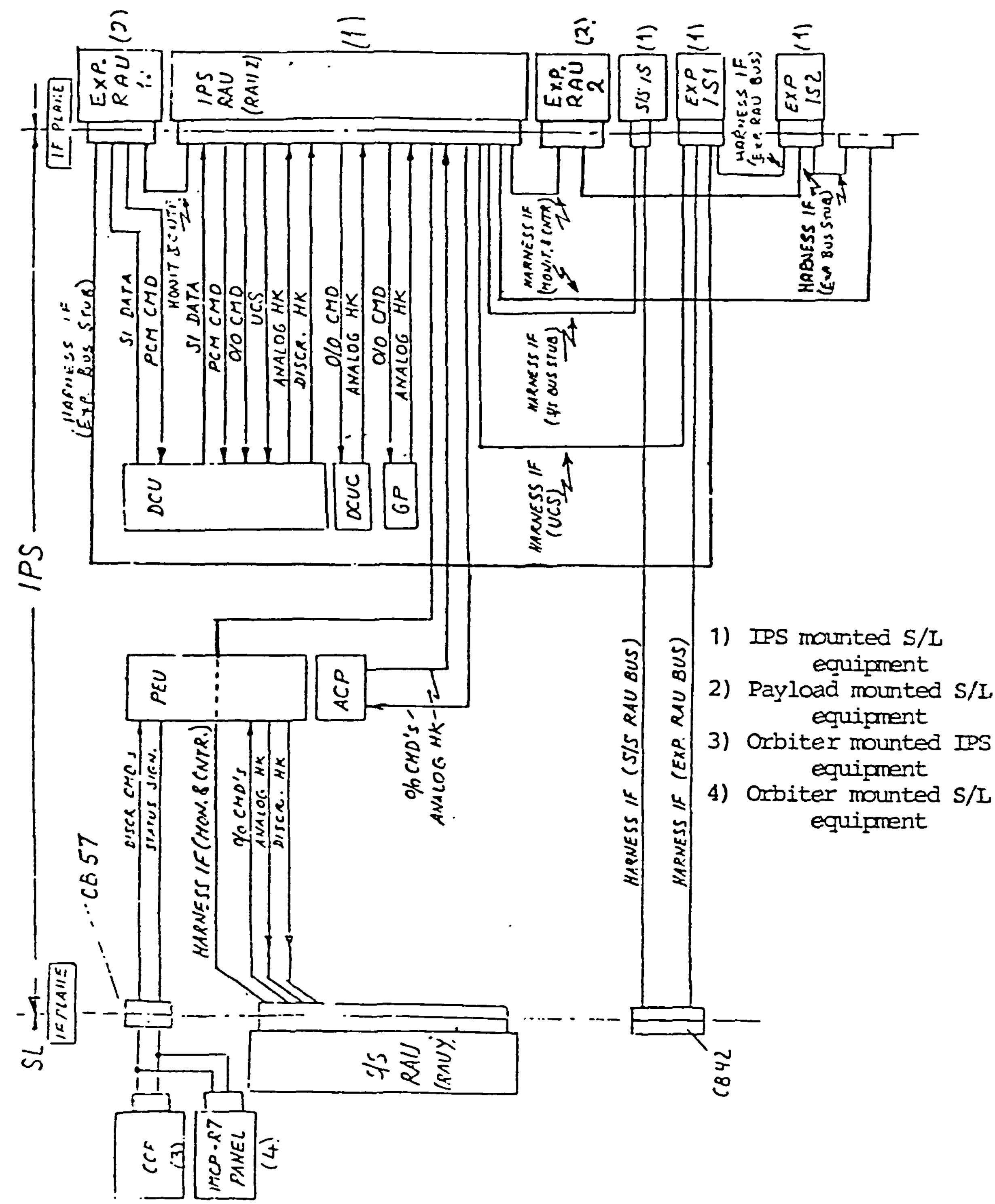


FIG. 2.1.3.3-4: IPS-SPACELAB SIGNAL INTERFACES

2.1.3.4 Software Interface

2.1.3.4.1 Interface with the Orbiter/STS

2.1.3.4.1.1 Uplink

2.1.3.4.1.1.1 Orbiter Data

The GN&C data provided by the Orbiter via SCOS to IPS will be in the format defined in Appendix A §A.1.1.2.28 and A.1.1.2.29 of ICD-02-05301; the data are updated at the rate described in § 3.4.1.3. of ICD-02-05301.

2.1.3.4.1.1.2 IPS Uplink Commands

By uplinking commands to the IPS via the Orbiter (MDM-link) the ground will have the same functional capability to control the IPS that the crew has using the keyboard.

IPS uplink commands can be subdivided into commands which are executed directly by the Subsystem Computer Operating System (SCOS) without IPS SW intervention, and commands executed by IPS SW.

Commands which are directly executable by SCOS are beyond the scope of this document.

The number of uplink commands which are to be executed by the IPS SW will be one for every ITEM in each IPS FFD, with the exception of IEL and IME FFD's, plus one command for each of the Function Keys (FK's) dedicated to the control of the IPS (however, some of the commands will never be used via uplink).

IEL and IME FFD's only contain items belonging to the category of commands directly executable by SCOS.

2.1.3.4.1.1.3 Formats of IPS SW Data Files on MMU

The following data files are used by IPS SW:

1) Objectives (File OBJECT)

2) Filter Gains (File AGAINS)

3) OSP Data (File LORDFL)

4) Mission Dependent Parameters (File COMPAR)

5) DCU Fast Loop SW (File DCUFAS)

6) DCU Self Test SW (File DCUTES)

7) Monitor Parameters (File MONLIM)

8) TCA Parameters (File TCALIM)

Data files used by IPS SW will be stored on MMU as SCOS "User Files". The general format of an SCOS User File is described in MA-MA-0075, para 5.4.2.3.

2.1.3.4.1.2 Downlink

2.1.3.4.1.2.1 IPS Data in TMB

IPS data downlinked as part of the Telemetry Buffer (TMB) are grouped into downlink frequency subparts. Within each frequency sub-part, data are grouped into blocks of consecutive words and for each word the ID#'s of the items allocated to that word are listed.

The data are located in the TMB in the order given below:

ANALOG adjacent ID numbers are located in one 16 bit word

WORD 1	WORD 2	WORD 3	WORD 4
WORD 5	WORD 6	WORD 7	WORD 8

DISCRETE 2 rows of ID numbers are contained in one 16 bit word

bit 1	bit 2	bit 3	bit 4	bit 5	bit 6	bit 7	bit 8
bit 9	bit 10	bit 11	bit 12	bit 13	bit 14	bit 15	bit 16

For fields not filled with ID numbers the following applies:

☐	(blank):	bytes or bits not contained in TMB
SL	:	bytes or bits already acquired by SL, respective words not accounted for in IPS TMB (see IF-IS-0001, para 4.2.1.3)
x	:	bytes or bits not available for IPS, but respective words accounted for in IPS TMB
--	:	bytes or bits available as IPS spares, accounted for in IPS TMB
IN	:	invalid bytes or bits due to 20 Hz acquisition, accounted for in IPS TMB

2.1.3.4.2 Interfaces with Spacelab

2.1.3.4.2.1 Interfaces with Spacelab SW

The Spacelab generated SW packages listed in SR-IS-0001, § 3.1.5.2, in the versions specified below in Table 2.1.3.4.2.1-1, are utilized for the production or for the on line support of IPS SW ("Version" defines the revision status).

SW Package	Version
- Host Macro Assembler (XMAS)	4.0
- Host Linkage Editor (XEDL)	4.0
- HAL/S-CII Compiler, HALLINK	6.01
- HAL/S-IBM Compiler	16.46
- Subsystem Computer Operating System (SCOS)	8.6
- System Generator (SYSGEN)	3.6
- FFD Skeleton Generator (SKLGEN)	3.6
- Memory Configuration Generator (MCTGEN)	3.6
- Flight Tape Generator (FLTGEN)	3.6
- Data Base Generation and Maintenance (DBGM)	3.11

Table 2.1.3.4.2.1-1: Spacelab SW Packages Versions

2.1.3.4.2.2 CDMS SW Resources Utilization

2.1.3.4.2.2.1 SSC SW Sizing

SSC core size utilization (budget and actual) is as follows:

SW Item		Words
a)	SSC SW without IPS	40000
b)	IPS SW (Cat. 1 and 2)	22500
c)	Second Fixed Format Display Buffer	1500
	TOTAL	64000
	available	65500

Item a) is defined as the SSC SW System as optimized to a SL-2 configuration.

Item b) is as specified in IF-IS-0001, § 5.2.1.

Item c), actually operations dependent, is required when it is expected to operate two DDS's concurrently with the SSC.

2.1.3.4.2.2.2 SSC SW Timing

SSC CPU load and access budget is as specified in IF-IS-0001, para 5.2.2.

Due to the interrupts allocation specified in IF-IS-0001, para 5.2.3 it is expected that it will not be possible to update two concurrent FFD's at the nominal rate of 1 Hz. The consequence is that the FFD update frequency must be set at SCOS generation time to a lower value which is consistent with the overall SSC SW load, in order to permit tasks running at priorities lower than the FFD-updating task to access the CPU and in order to avoid the generation of repetive SOE's 8A01 (Overrun of the FFD updating task).

A nominal FFD update frequency for an IPS mission is established to be 0.75 Hz. However, as it is considered operationally acceptable that FFD updates could be slowed down to a minimum of 0.33 Hz, the CPU margin obtainable by reducing the "nominal IPS FFD update frequency" to this value is the current CPU reserve. Reduction to a minimum of 0.5 Hz is controlled by ESA, for IPS SSC SW and operation contingencies. Lower update frequency values need joint approvals from ESA and the organization responsible for the SL SW maintenance.

2.1.4 IPS Performance

The IPS performance has been analyzed using a finite element based simulation model incorporating the following substructures:

Orbiter from NASA/MDTSCO

Pallet from ESA/ERNO

IPS from DORNIER

Payload from DORNIER

The overall structure is modelled to be free-free.

The following data apply to the Orbiter finite element model (payload doors open):

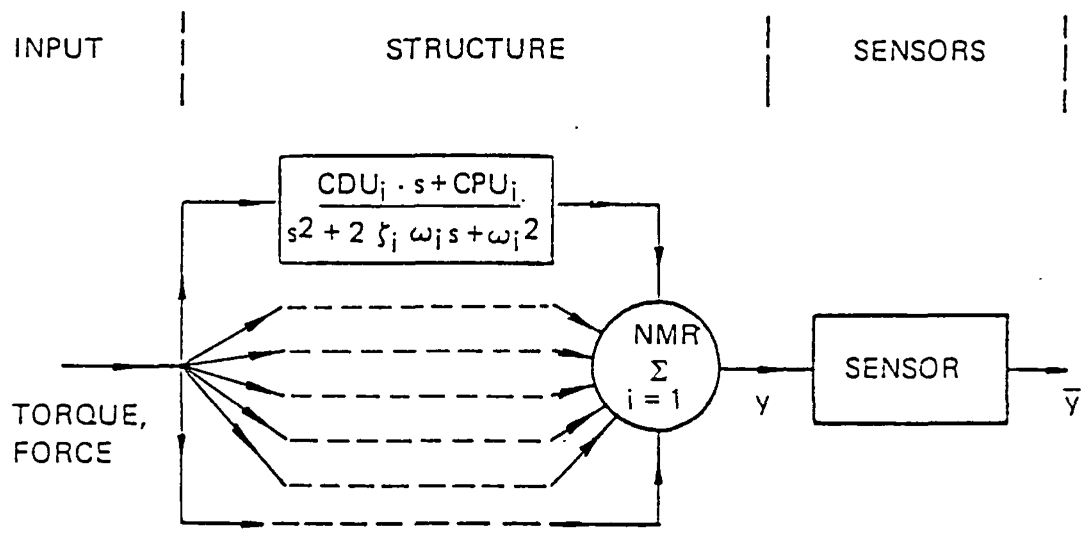
- Mass $m = 81074$ kg
- MOI $J_{xx} = 1.2 \cdot 10^6 \cdot \text{kgm}^2$

  $J_{yy} = 8.8 \cdot 10^6\ \text{kgm}^2$

  $J_{zz} = 9.1 \cdot 10^6\ \text{kgm}^2$

  $J_{xz} = 0.3 \cdot 10^6\ \text{kgm}^2$

The basis of the overall structural model is a system representation using parallel second order oscillators between control and disturbance inputs and sensor outputs as in figure 2.1.4-1.



NMR	:	number of retained modes
y	:	displacement at sensor station
$\bar{y}$	:	measured displacement at sensor station
CPU_i	:	proportional coupling coefficient (for i'th mode)
CDU_i	:	derivative coupling coefficient (for i'th mode)
ζ_i	:	damping coefficient (for i'th mode)
ω_i	:	modal frequency (for i'th mode)

Figure 2.1.4-1: Parallel Oscillator Representation

For every IPS look angle and for every IPS payload an individual structural model is used.

External disturbance forces applied to this model are:

- VRCS thruster firing and
- Man motion.

Location of Orbiter Disturbance Sources
(Co-ordinates relative to the spacelab co-ordinate system in millimeters):

- Man Motion X = 8806 Y = 0 Z = 1198
- Vernier Thrusters

	X	Y	Z
1	3456	-1516	-1267
2	3456	1516	-1267
3	34968	-3807	1499
4	34968	3087	1499
5	34968	-2997	1408
6	34968	2997	1408

Magnitude of Orbiter Disturbances

- The man motion to be considered by the IPS design shall be as shown in Figure 2.1.4-2. This motion shall be applied in each axis individually.

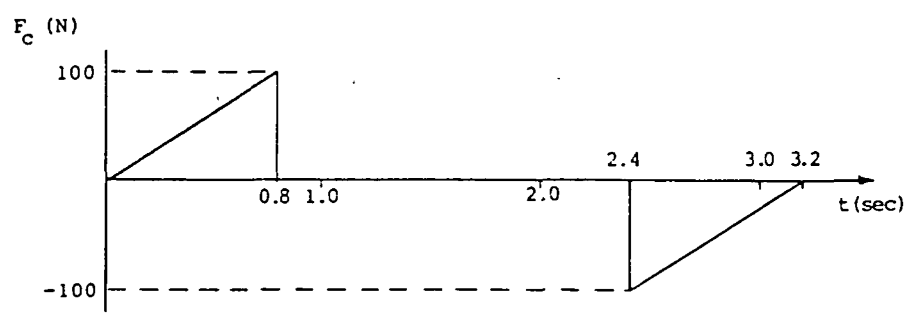


Figure 2.1.4-2: Man Motion Disturbance

- The IPS design takes into account the vernier thruster thrust levels and six firing combinations shown in table 2.1.4-1 for an 80 msec duration. The IPS design shall also take into account an Orbiter limit cycle motion of ± 0.1 degree.

Thruster No.	X	Y	Z	Remarks
1	-3.56	75.62	- 78.29	
2	-3.56	- 75.62	- 78.29	
3	0	106.8	- 2.67	
4	0	-106.8	- 2.67	
5	0	0	-106.8	
6	0	0	-106.8	
Thruster Combinations	1,3,5 2,4,6	1,2 5,6	1,4 2,3	+ ve rotation - ve rotation

Table 2.1.4-1: Thrust vectors (N) and firing Combinations

With these orbiter dynamics and FEM's for 200, 2000 and 7000 kg payloads the following performance values have been simulated (Table 2.1.4-2):

Payload Mass	Disturbance		Sim. Result (arcsec)
200 kg	Man Motion		
		Lat	3.2
		Roll	10.0
	Thruster Firing		
		Lat	5.6
		Roll	10.0
2000 kg	Man Motion		
		Lat	3.9
		Roll	4.0
	Thruster Firing		
		Lat	5.3
		Roll	5.1
	Quiescent Stab.		
		Lat	0.8
		Roll	3.0
7000 kg	Man Motion		
		Lat	6.0
		Roll	-

Table 2.1.4-2: IPS Performance

2.2 Position and Hold Mount (PHM)

2.2.1 General

The PHM is a small pointing facility for experiments weighting up to 200 kg and calling for low or medium stability. The pointing stability is based on the orbiter ($\pm$ 0.1 deg.).

After performing a feasibility study (Phase A), which included a two axes demonstration model, Dornier System has completed the definition of the PHM (Phase B) in October 1982.

2.2.2 Technical Concept

The Position and Hold Mount (PHM) is a two axis pointing facility for smaller Spacelab payloads or payload clusters of up to 200 kg mass (see Fig. 2.2.2-1, 2.2.2-2). Its elevation-over-azimuth two axes gimbal assembly provides up to 360 degrees of freedom range in azimuth and up to 180 degrees of freedom range in elevation. The medium pointing stability based on the stability of the Orbiter is $\pm$ 0.1 degrees.

As a Spacelab subsystem the PHM relies on Spacelab/Orbiter support in the areas of data management (CDMS), power supply, thermal control services for electronics boxes (cold plates) and attitude information (IMU). It was one of the driving design goals to ease the use of these Spacelab/Orbiter services for the potential PHM user and to make interfaces as simple, reliable, and modular as possible to cover a broad application spectrum for the PHM payloads.

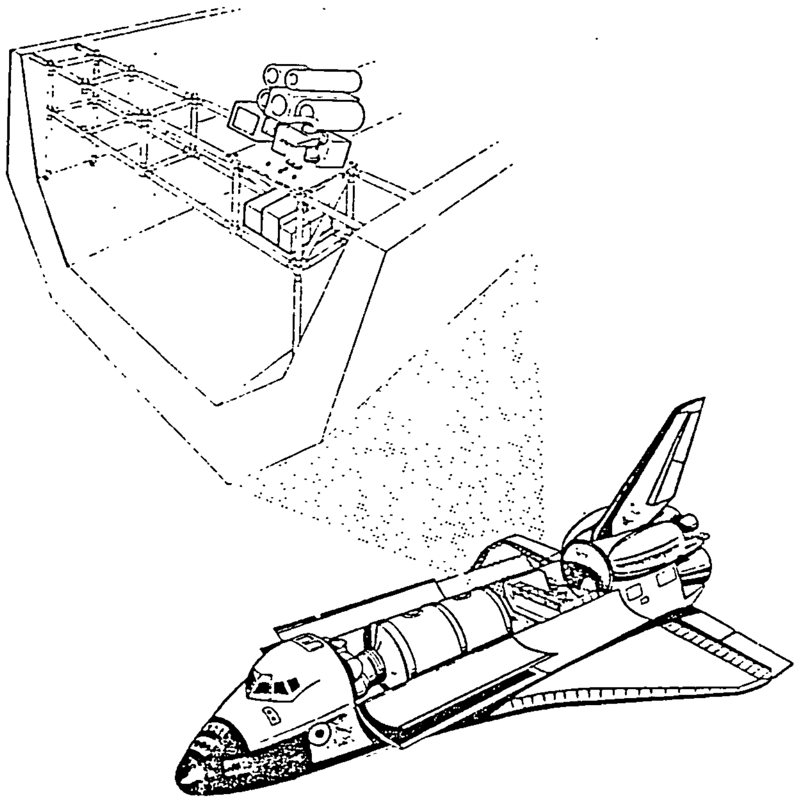


FIG. 2.2.2-1: PHM WITH TYPICAL PAYLOAD CLUSTER

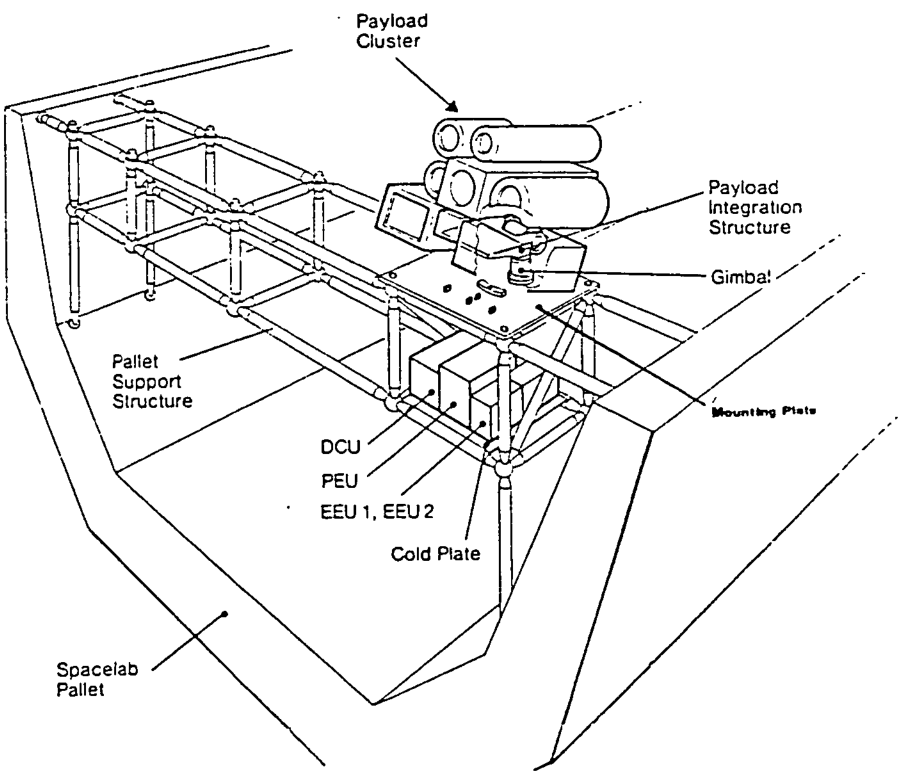


FIG. 2.2.2-2: PHM WITH TYPICAL PAYLOAD INTEGRATED ON A SPACELAB PALLET SUPPORT STRUCTURE

The PHM Spacelab subsystem consists of the following major assemblies:

- The Gimbal Assembly (GA) with the Drive Mechanism Subassembly (DMSA) for rotating the payload, the Clamping Mechanism Subassembly (CMSA) for securing the payload during launch and landing and the Structural Elements Subassembly (SESA) for overall stiffness.

- The Electronics Assembly (EA) with its Power Electronics Unit (PEU) for power supply and the Data Control Unit (DCU) for data and operation management. The Emergency Electronics Units (EEUs) are special electronic units for contingency back up operating modes.

- The Ground Support Equipment (GSE), partitioned in the Electrical Ground Support Equipment (EGSE) and the Mechanical Ground Support Equipment (MGSE), which both supply the necessary checkout and verification tools.

Special attention throughout the whole PHM design was payed to the safety concept and its mechanical and electrical implementation. The safety elements have to guarantee the integrity of the PHM and its payload during any mode of operation, especially during launch of the Shuttle, in orbit operations and during descent and landing of the Orbiter. Safety is among others achieved by a completely redundant retraction and stowing capability for the payload.

This safety aspect was one of the design drivers for the overall PHM technical concept.

Due to its modular design the operating capabilities and performance of the basic PHM design as described in this document can be extended relatively easy by adding attitude sensors (e.g. sun sensor), the necessary sensor couplers and software in the DCU.

This aspect was not covered in this phase B study, but experience with the PHM demonstration model (PHM phase A-study) indicated with a Dornier supplied sun-sensors, that accuracy improvement by a factor of 5 is easily achieveable.

As the PHM is conceived to cover a broad range of applications, some mission dependent hardware has to be taylored according to the specific payload requirements and has to be supplied by the user. These are the following elements:

- a Payload Integration Structure (PIS) as linking element between payload and PHM
- a Mounting Plate (MP) as linking element between PHM and the Spacelab Pallet support structure
- a mission tailored harness for linking PHM and payload elements to the electronics
- a thermal protection for the PHM elements and for the payload according to the thermal requirement of the particular mission
- some structural elements to optimize the load paths from Spacelab to the payload.

2.2.2.1 Mechanical Concept

The major design drivers leading to the mechanical design concept are the requirements:

- for accommodation of payloads of various shapes and sizes,
- for a gimbal freedom permitting pointing ranges of any sector within a total of a hemisphere,
- for the ability to secure payload and gimbals for safe return under all circumstances, and
- for the flexibility of mounting the PHM at various places and in any direction on the Spacelab pallet by means of mission dependent support structures.

Thus, the Gimbal Assembly (see Fig. 2.2.2.1-1) fulfils the mechanical payload accommodation and safety requirements, given in short form:

- payload mass: 200 kg (including Payload Integration Structure)
- payload moment of inertia: $50 \leq J \leq 500$ kg/m^2 (incl. PIS)
- payload centre of gravity relative to PHM coordinate system for

  side mounted payloads $Y_{max} \leq 625$ mm

  end mounted payloads $X_{max} \leq 1125$ mm

- payload service
  - o securing of payload during ascent and descent.

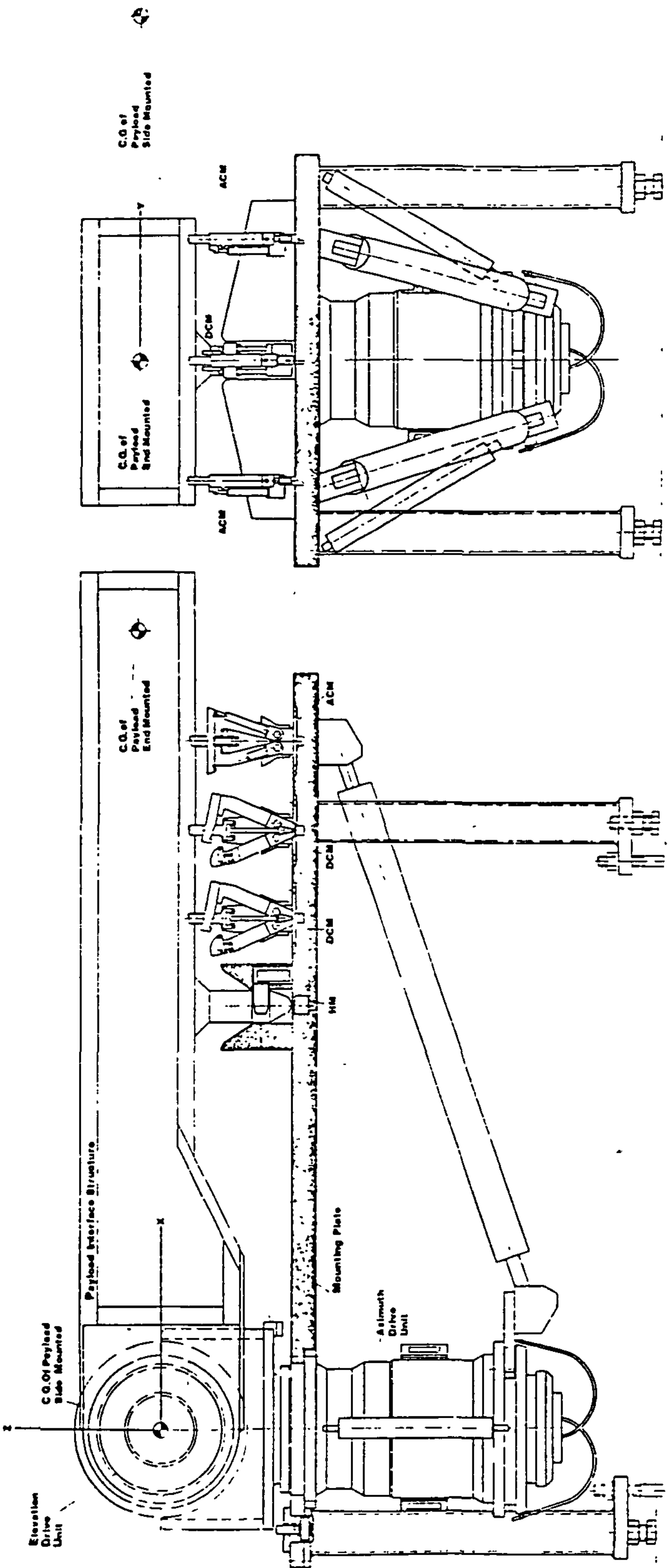


FIG. 2.2.2.1-1: PHM GIMBAL ASSEMBLY

- position keeping at zero power by means of brakes (Orbiter vernier thrusters only)
- safety requirements
  - o structural integrity to be guaranteed by gimbal assembly and clamping mechanism
  - o payload not to penetrate the dynamic envelope of the Orbiter and not to damage surrounding equipment or experiments.

The Gimbal Assembly (see Fig. 2.2.2.1-2) covers the following subassemblies:

- The Drive Mechanism Subassembly (DMSA) has the task to rotate the payload in the desired direction or to perform the commanded motions like scanning or tracking. Main elements within the DMSA are the Drive Units (DU) (see Fig. 2.2.2.1-3) for the azimuth (ADU) and elevation (EDU) axes. They contain the DC-motors for torque generation, the resolvers for relative angle measurement, the brakes to hold the payload in any direction, the position indicators as a backup to enable emergency retraction and the bearings to take the load. The Yoke links ADU and EDU together and carries auxiliary items like connector brackets etc.

- The Structural Elements Subassembly (SESA) consists of the user supplied Payload Integration Structure and Mounting Plate. Furthermore the struts as linking element between the ADU and the MP supply the necessary stiffness for the Gimbal Assembly. The adjustable mechanical endstops may limit the range of the payload and serve as ultimate safety device in case of malfunction.

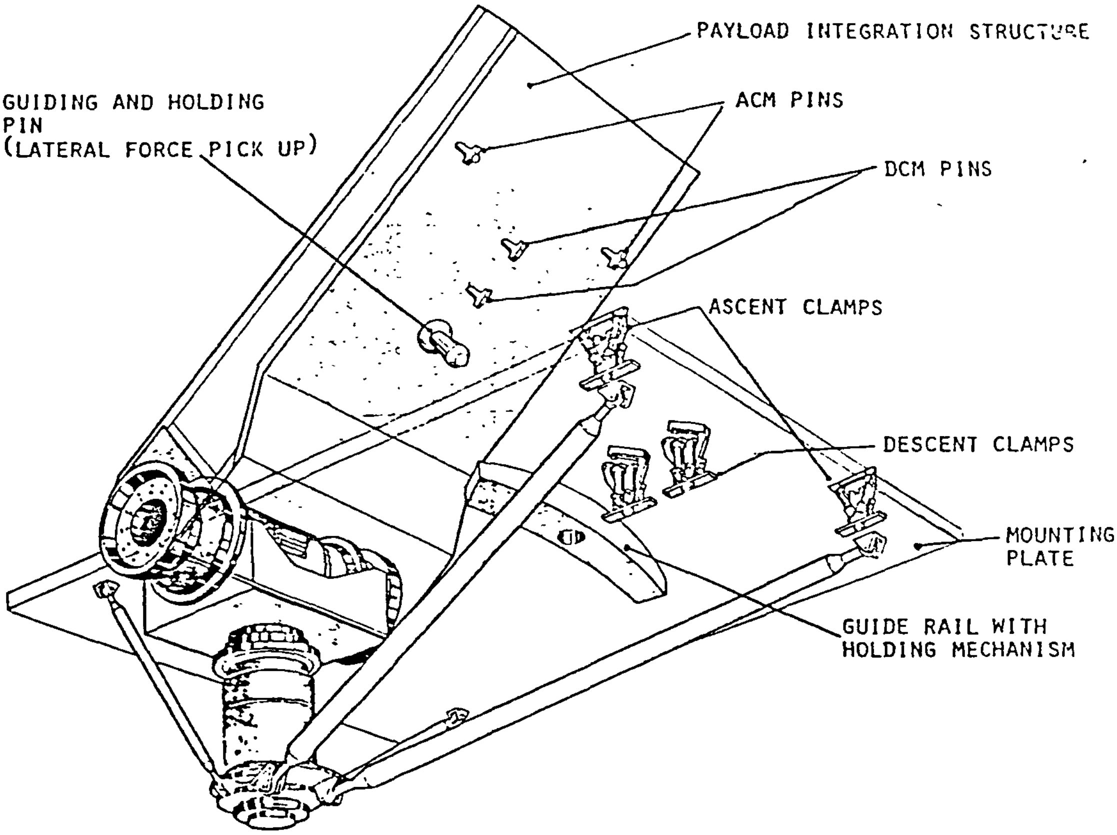
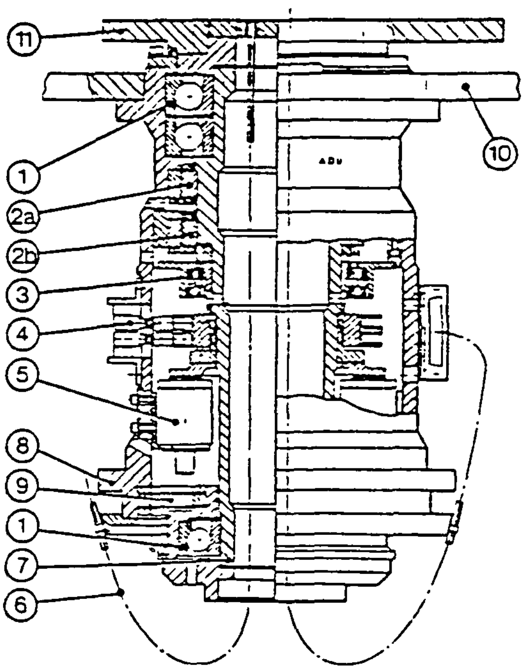


FIG. 2.2.2.1-2: PHM GIMBAL SUBSYSTEM AND CLAMPING MECHANISM



The Components of the Drive Units are:

1 Ball Bearings
2a Motor for nominal operations
2b Motor for emergency operations
3 Resolver
4 Position Indicators
5 Solenoid Brakes
6 Flexible Wire Harness
7 Shaft
8 Housing
9 Membrane
10 Mounting Plate } Mechanical Interface
11 Yoke

Housing and shaft are one piece components for maximum stiffness.

FIG. 2.2.2.1-3 PHM DRIVE UNIT COMPONENTS

- The Clamping Mechanism Subassembly (CMSA) design features different elements for different tasks. The Ascent Clamping Mechanism (ACM) takes the loads imposed on the PHM/payload ensemble during launch of the Space Shuttle. The Descent Clamping Mechanism (DCM) is responsible for securing of PHM/payload during Orbiter descent and landing. Main actuating element in both units are memory metals which, subjected to heat, change their physical dimensions, thus actuating the clamp.

  The Holding Mechanism serves as a mechanical guide for the zero-clamping-position and as intermediate stowage clamp during Orbiter RCS main thruster firing.

2.2.2.2 Thermal Concept

The thermal concept of the PHM requires to cover the Spacelab so'called "hot case" and "cold case" operational conditions.

As it is not possible, to cover all possible user and payload requirements each payload + PHM thermal concept must be individually tailored.

The PHM reserves nevertheless within its PEU some power, dedicated for thermal control, as well as the software is flexible enough to incorporate even complex thermal control switching functions.

It is assumed that whatever thermal concept will be chosen for the PHM and its payload, it is as much as possible of the "passive thermal control" concept.

2.2.2.3 Electrical Concept

The Electronics Assembly of the PHM is designed for operation via the Spacelab CDMS in terms of data acquisition, data transmission, operations service, etc. and for supply by the Spacelab power system in terms of main- and essential power.

The electronics concept is based on the following interfaces, shown also in the interface blockdiagram (fig. 2.2.2.3-1):

- PHM-SSC link for data exchange with the CDMS by an IPS-type of RAU (which is an EXP-RAU)
- PHM-EXC link for data exchange with the payload by an EXP-RAU
- PHM main power bus supply by Spacelab EPDB for normal operations
- PHM emergency retraction and stowage commanded and controlled from Orbiter AFD panel R-7, via bracket 57.

Three different electronics units will control/perform the PHM operations/functions as follows:

- the Data Control Unit (DCU) will handle all data traffic and software duties imposed on the PHM by operations requirements
- the Power Electronics Unit (PEU) will supply all necessary power and switching functions for the PHM, except the emergency functions
- the Emergency Electronics Units (EEU), which is present in redundant form as EEU-1 and EEU-2, will serve as power supply, command receiver, and control signal generator in case of emergency operation.

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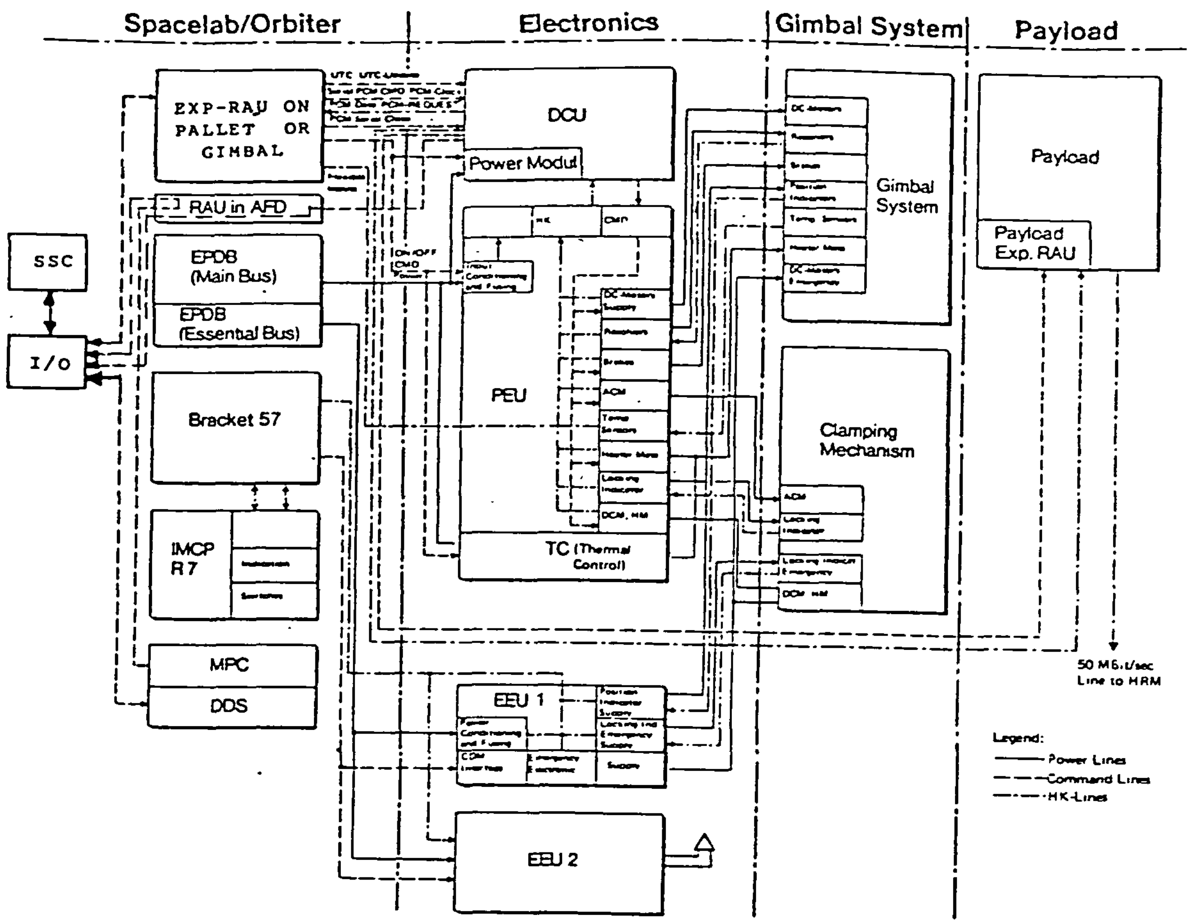


FIG. 2.2.2.3-1: ELECTRONICS ASSEMBLY

The DCU, as most complex unit of the Electronics Assembly, is described in the following in more detail:

The DCU handles and synchronizes the data traffic between the PHM and the Spacelab CDMS (Subsystem Computer, Experiment Computer) via RAU.

The dataprocessor 430-R controls the traffic on the MUDAS dataway and is able to perform logic and arithmetic operations. This capability is used for calculation of the attitude measurement and control algorithms, and for operating the PHM according to the PHM modes and their routines. The programs of the data-processor are stored in the memories, which are of PROM and RAM type.

The DCU is built to the Dornier-MUDAS space standard. It consists of functional modules, which are connected to the MUDAS dataway. The modules are controlled by the data-processor, which besides the control function is able to perform logic and arithmetic operations.

2.2.2.4 Software Concept

The PHM software concept is governed by the rule to cover as many operational and software tasks in the PHM dedicated processor as possible, to make the PHM a real self standing, autonomous pointing facility to ease the usage and to facilitate the testability.

Nevertheless it is clear, that the offered basic Spacelab Subsystem Computer (SSC), Data Display System (DDS) services are used.

That leads to a distribution of the PHM software in the PHM DCU, covering all operational sequencing and pointing control functions and in the SSC for usage of basic CDMS software services. Additionally the S/W concept is so flexible to incorporate easily additional sensors to increase the pointing accuracy (for example: sun sensor).

2.2.2.4.1 DCU Software

The DCU software covers the following topics:

- Controller S/W
- Timing Control and Organization S/W
- Housekeeping S/W
- PHM-routines and subroutines S/W
- Transformation, and calibration S/W

2.2.2.4.2 CDMS Software

The main task of the CDMS software for PHM is to transfer data, as shown in Fig. 2.2.2.4.2-1.

Inputs are acquired from

- MDM: commands and orbiter attitude information
- Keyboard: commands
- RAU: values to monitor or to display and hand controller information

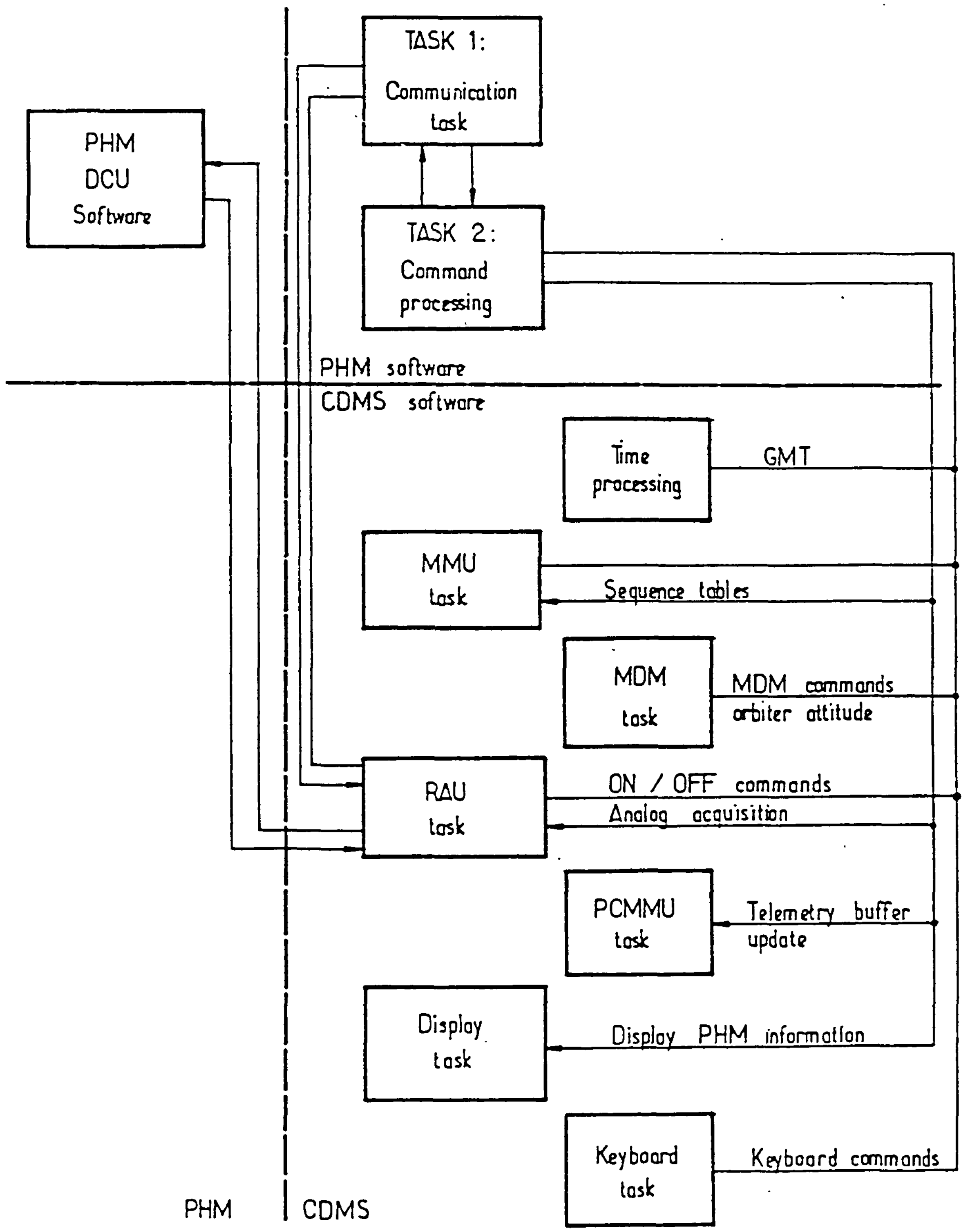


FIG. 2.2.2.4.2-1: CDMS SOFTWARE BLOCK DIAGRAM

- MMU: command sequences
- MTU: time information

Outputs are routed to

- Display: measurement points, monitoring status error messages, etc.
- RAU: ON/OFF commands and serial output
- MMU: update command sequence and user files
- PCMMU: Error messages, mesaurements point values, CDMS hardware status, etc.

SCOS is in charge to access the different peripherals but some application software has to be provided to initialize the different transfers.

The proposed solution for PHM CDMS software is based on a two tasks structure

- The first task will have to handle dialog with the PHM via the RAU serial channels

- The other task will have to handle all other transfers required by the PHM utilization.

For both tasks the application language will be HAL/S.

2.2.2.5 Operations Concept

The operations concept of the PHM (see Fig. 2.2.2.5-1) is based on one side on the CDMS-subsystem computer (SSC) and on the other side on the PHM-DCU processor (MUDAS).

These two processors control together the operations of the PHM.

Four major PHM modes cover the operating requirements and give individual payload operating flexibility.

- Individual command mode (ICM)
- Manual Pointing Controller mode (MPC)
- Timeline Program Mode (TPM)
- Power Down Mode (PDM).

Included in all modes is an "alert trigger", which indicates PHM malfunctions, to alert the Spacelab crew (FFD-SSC message line).

The Individual Command Mode reflects the necessity, mainly for safety reasons, to have a step-by-step command possibility, initiated and controlled by CDMS-keyboard. Within the ICM the following submodes are possible:

- the ICM-end item-mode (ICM-EIM)
- the ICM-PHM-routine-mode (ICM-PRM).

The ICM-EIM is characterized, that by CDMS command, a list of PHM-end items may be commanded and controlled.

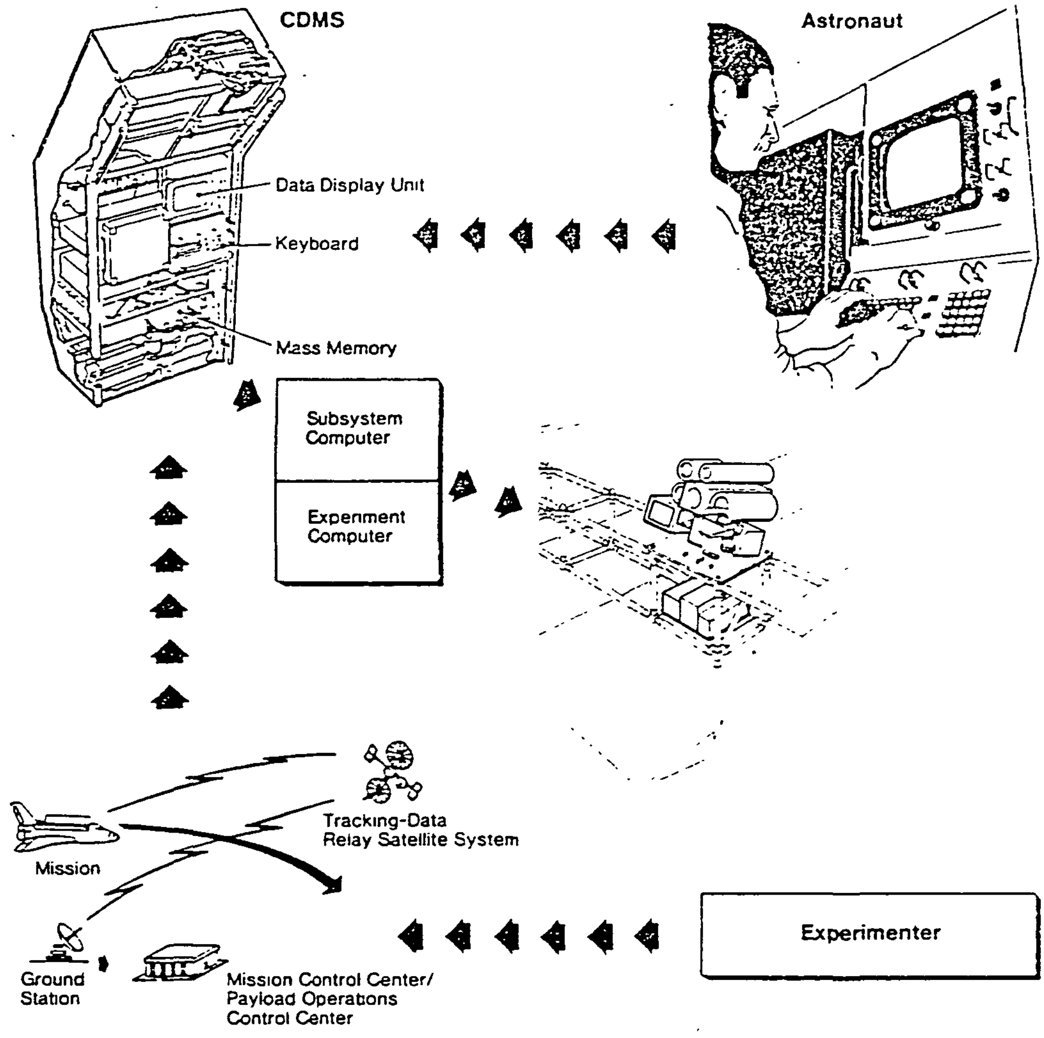


FIG. 2.2.2.5-1: COMMAND LINES FOR PHM OPERATIONS

End items are:

- DC-motors
- clamps
- brakes
- retraction bolt
- resolvers
- thermal elements

The ICM-PRM is characterized, that by CDMS command, individual PHM-routines may be initiated by a GO-command and stopped arbitrarily by a STOP-command, not dependent on GMT.

The Timeline Program Mode is an automatic, time dependent mode which is preprogrammed by the appropriate PHM-routines and their characterizing parameters.

The TPM may be operated from:

- the CDMS keyboard by the on-board crew
- the MMU by initiation of the on-board crew
- the POCC to make corrective actions or change the contents of the MMU.

The sequencing element operating the PHM in the TPM is a "Sequence Table" where the various PHM routines are expressed by their associated parameters. The "Sequence Table" (S.T.) shall be initiated by a GMT start-time and ended by a GMT end time. The S.T., may be constituted of several Sequence Table blocks. One block shall comprise one specific parameter set.

To have enough flexibility within the preprogrammed routines, a "Sequence Table Change Procedure" permits fast onboard modifications by the Spacelab crew.

The following PHM routines are defined:

- Activation and Initialization Routine (AAIR)
- Control Parameter Loading Routine (CPLR)
- Relocking Routing (RELR)
- Slewing And Holding Routine (SAHR)
- Positioning And Holding Routine (PAHR)
- Tracking Routine (TRAR)
- Scanning Routine (SCAR)
- Bias Controller Routine (BICR)
- Deactivation Routine (DEAR)

The Manual Pointing Controller Mode shall enable the on-board crew of manual pointing and slewing the PHM. The MPC-commands will be treated as bias commands. The operating post of this mode is a Keyboard/joystic device in the AFD to produce the bias commands. After having selected an adequate PHM-routine, motion-characterizing parameters of this routine will be regarded as dummies and overwritten by the bias commands which are:

- Azimuth angular rate
- Elevation angular rate.

The Power Down Mode is introduced to keep the PHM thermally controlled without switching on the PHM processor. The PHM is thus a purely CDMS operated mode concerning the commands as well as the feedback data.

For temperature measurements and control, the identical items are used as for the normal PHM thermal control.

The power will be drawn from a special power supply within the PEU.

2.2.2.6 Safety Concept

The PHM safety concept is based on the following design requirements:

- the PHM and its payload shall be non-jettisonable devices
- the PHM and its payload shall require no EVA - operating or a back up to come to a safe configuration
- the payload shall be fixed to the PHM during launch, in-orbit-operations, and descent/landing.

The safety concept thus is most concerned with the structural integrity of the PHM and its payload. To guarantee this, the PHM/payload combination requires through all high stress phases, a maximum strength and stiffness of the mechanical configuration which is only met, when proper clamping is achieved.

Thus, the safety concept is mainly a problem of guaranteed clamping of the PHM through ascent and descent of the Space Shuttle/Orbiter. As clamping can be visually verified before launch, the main concern is with the clamping prior to descent. This is treated in the following:

If normal flight mode retraction and clamping has failed (even in case of CDMS failure) or if normal flight mode is no longer possible, an emergency retraction and clamping mode will be performed which relies on fully redundant hardware components, separate command, monitoring, and power supply lines.

Emergency clamping thus is performed manually within the following scheme:

- a redundant set of electrical and electro-mechanical items, will rotate the PHM axes to null position, independently of CDMS and DCU
- the emergency power electronics (there are two for additional redundancy), will power the above mentioned items. Power will be drawn from essential power bus
- the redundant clamping mechanism and its actuating elements will secure PHM/payload
- redundant end switched will indicate safe locking
- initiation, control, and completion of the emergency retration will be done via Orbiter Panel R-7, completely independent from CDMS command lines.

The announcement to the crew, that PHM requires an emergency retraction, will be performed by SCOS services to DDS, based on PHM housekeeping data.

The PHM mechanical safety devices are shown in figure 2.2.2.6-1.

The emergency retraction mode will be checked out for proper function after launch prior to start of the PHM's scientific mission.

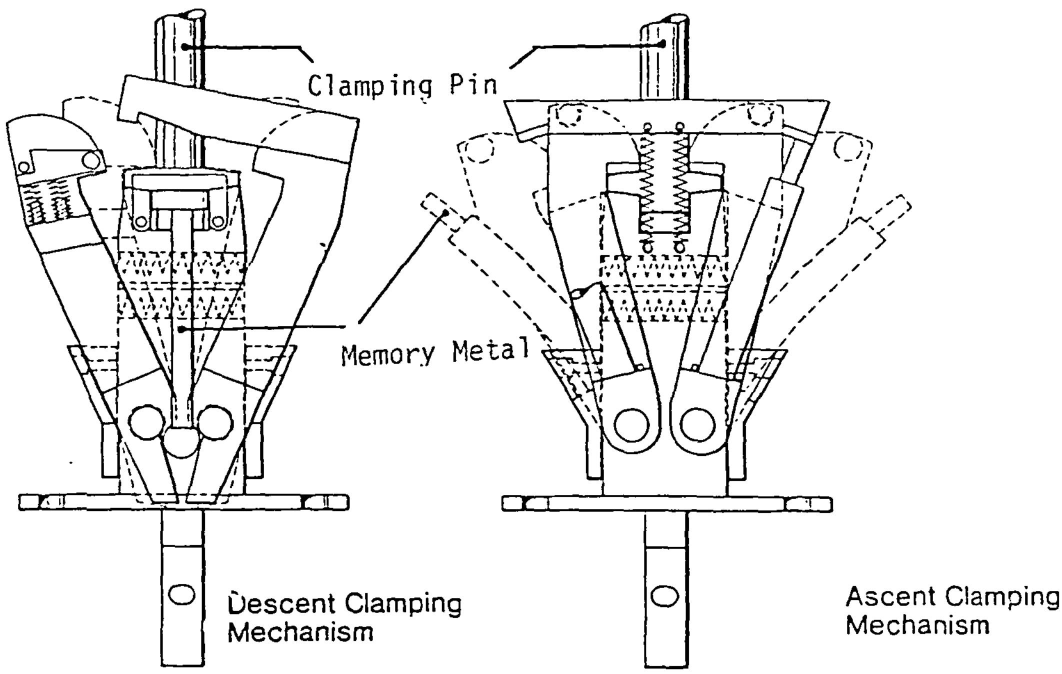


Fig. 2.2.2.6-1: PHM Safety Devices

2.2.3 Interfaces

The PHM will be integrated into Spacelab. This section describes the relevant Spacelab and Orbiter interfaces of PHM.

2.2.3.1 Mechanical Interface to Spacelab

- PHM/Payload Dynamic Envelope	not to exceed the Orbiter dynamic envelope and not to hit any surrounding Spacelab payload
- Total mass of PHM	m_{PHM} = 95 kg
o Mass of Gimbal Assembly	m_{GS} = 75 kg (incl. clamps)
o Mass of Electronic Assembly	m_{ES} = 20 kg
- Total mass of PHM and Payload	m_{TOT} = 295 kg (max.)
- Mounting interface for	
o Gimbal	- Azimuth Drive Unit flange, mountable to the user supplied Mounting Plate - Struts - Clamping Mechanism flange to be mounted to the Mounting Plate - Clamping Pins, to be mounted to the user supplied Payload Integration Structure - Endstops, according to PHM/Payload Dynamic Envelope
- Electronic Assembly	Common baseplate to be mounted to a Spacelab cold plate
- Mounting Plate	To be mounted to Mission dependent Spacelab Pallet Support Structure

- Mounting direction — any direction relative to the Spacelab coordinate system

- Coupled analysis — coupled analysis necessary for the combined Payload/PHM/Support Structure

2.2.3.2 Thermal Interface to Spacelab

As stated in 2.2.2.2 "Thermal Concept" there is at the moment no specific thermal interface to the Spacelab. But one can easily forsee, that the PHM would require thermal services from the Spacelab in the field of

- space and heat rejection capability for the power and data electronics by means of using a total or a portion of a Spacelab coldplate.

- space and heat rejection/injection capabilities for the Drive Mechanism Subassembly to comply with the "passive thermal control" concept. Space is needed in this case for a heat pipe radiator surface mounted to the cold plate.

2.2.3.3 Electrical Interface

To Spacelab:

- Data Control Unit

  Connected with one Connector to Spacelab Main Power Bus
  Connected to an EXP-RAU

- Power Electronics Unit

  connected with one connector to Spacelab Main Power Bus

- Emergency Electronic Unit

  EEU1 and EEU2 connected individually to Spacelab Essential Power Bus

- PHM Power Requirements

  180 W mean value, 2 axes
  280 W peak value, 2 axes
  150 W emergency, 1 axis

To Orbiter:

- AFD R7 Panel

  hardwired lines to PHM Emergency Electronic Unit via Bracket 57

2.2.3.4 Software Interface

- Payload to

  Experiment Computer Operating System (ECOS-S/W)

- PHM to

  Subsystem Computer Operating System (SCOS-S/W)

2.3 Antenna Pointing Mechanism

2.3.1 General

The Antenna Pointing Mechanism (APM) is the coupling/discoupling member between, for example, a heavy satellite and its spot beam antenna reflector, as shown in Fig. 2.3.1-1. It is specially designed for precise pointing within a limited pointing range. It incorporates within its cardanic suspension direct drive motors and precision angle pick-offs, controlled by its specially tailored electronics.

The development status is as follows:

Mechanism model built:

- Vibration Model
- Engineering Model
- Qualification Model
- Antenna Deployment and Pointing Mechanism (Fig. 2.3.1-1)

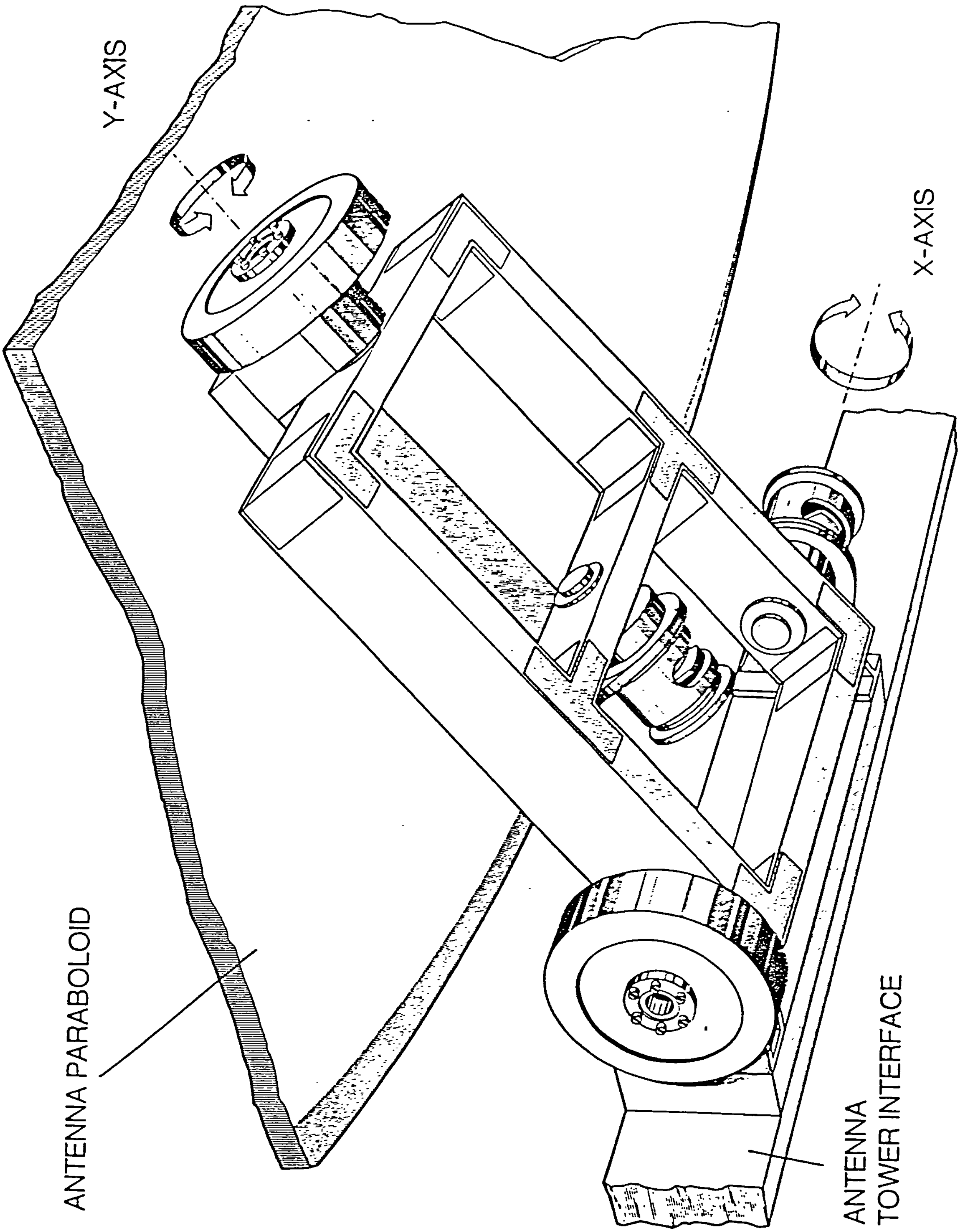


FIG. 2.3.1-1: ANTENNA POINTING MECHANISM (APM)

The Qualification Model survived successfully sine vibration inputs up to 20 g and interfacial temperatures from -70°C to +65°C during qualification tests. An accelerated life test programme at ESTL/England was passed successfully.

For the APM-Electronic a Breadbroad Model was followed by a Prototype Model which was manufactured to flight standard and has passed successfully electric performance tests.

2.3.2 Technical Concept

The technical concept aims at performing the following two pointing tasks:

1. precision pointing of an antenna as required by the WARC regulations for direct satellite broadcasting with a pointing accuracy of 0.01 degrees for the mechanism

2. antenna beam shift from one country to another (repointing)

The APM development at Dornier led to the following measured performance data:

Main Performance Data

- Antenna Inertias up to 18 kgm^2
- Deployment Range up to 180 degree
- Fine Pointing Range +/- 1,5 degree
- Pointing Accuracy +/- 0,01 degree
- Dynamic Performance above 2 Hz

- Speed above 0,3 degree/sec.
- Mass 3,5 kg
- Thermostable CFP (Carbon Fibre Plastic) structure
- Temperature Range -160^{o}C up to +120^{o}C
- Vibration Level up to +/- 50 g

2.3.2.1 Mechanical Concept

The APM is of modular design. Drives, angle pick-offs, bearings are the same on both axes. The frames of CFP material can be adapted to the specific interface requirements of both antenna and spacecraft body.

Basically the framework corresponds to a cardanic suspension. Its axes arrangement can be built centrally or shifted apart to allow deployment actions for the antenna from the launch to an operational condition.

Each axis has a direct driving stepper motor and a resolver as angle pick-off. The bearings are dry lubricated and therefore free of maintenance and free of backlash as well.

If required the APM can be equipped with a clamping device for direct load transfer during launch. A relocking device may be implemented to fix the antenna in a central position upon command.

2.3.2.2 Thermal Concept

The mechanism's heat input is low enough not to require an active thermal control.

The used CFP material has good thermal stability.

2.3.2.3 Electrical Concept

APM Electronic

It includes all the necessary electronic equipment of the Antenna Pointing Mechanism. The APM-Electronic controls the drives, feeds the angle pick-offs which are precise resolvers, processes their output signals into digital format, accepts commands and generates status signals. The input and output signals of the APM-Electronic are adaptable according to various satellite interface requirements. Each of the two pointing axes is connected to a completely redundant set of electronic circuits.

Technical Data of the APM-Electronic

- Power Input	15 W
- Mass	3.9 kg
- Overall Dimensions	363 mm long
	166 mm wide
	145 mm high

2.3.2.4 Software Concept

Not applicable

2.3.2.5 Operations Concept

The APM can be operated in the following modes:

- open loop

  specific commands related to effective antenna beam direction

- closed loop

  control loop using the angle pick-offs as position measuring devices

- closed loop with RF sensor

  control loop using an RF sensor as position measuring device located apart from the mechanism and sensing the effective antenna beam direction.

  Introduction of individual bias settings resulting from on-orbit calibration of the antenna performance poses no problem.

The APM is able to perform continuous antenna pointing in order to counteract residual satellite nutations over a life time of ten years.

Its dynamic capability corresponds to a bandwidth above 2 Hz.

The APM's stiffness allows for complete antenna pointing performance testing and calibration on ground with gravitation influence.

2.3.2.6 Safety Concept

The mechanism incorporates complete redundancy of motors and angle pick-offs. The electronic is completely redundant as well.

The relocking capability adds further reliability, which means, that in case of any failure within the drive branches the mechanism can be relocked to its central position by means of its telecommanded Relocking Device.

The life tests have proved the APM's reliable performence under qualification conditions. Special bearing tests confirmed their life endurance.

2.3.3 Interfaces

Mechanical I/F

Both the antenna frame and the ground plate can be adapted to special versions of reflector dish and satellite structure.

Electrical I/F

Mechanism and APM Electronics are to be interconnected by cable lines. Adaptation of the Electronics to the S/C power bus poses no problem.

Thermal I/F

Thermal shielding of the mechanism should be foreseen according to the specific implementation conditions.

Vibrational Dynamic I/F

Depending on the S/C structure and antenna characteristics the APM's stiffness allows for loads up to 70 g.

Operational Dynamic I/F

The APM provides a high dynamic bandwidth and high inherent damping. Therefore the matching of the dynamic behaviour of the APM to specific satellite requirements poses no problems.

3. REQUIREMENTS ANALYSIS FOR SPACE STATION POINTING SYSTEMS

3.1 General

In this section the mission models are analyzed with respect to pointing system aspects. The following relevant mission areas have been identified

- Astrophysics
- Earth and Planetary Remote Sensing
- Environmental Observations
- Communications
- Life Sciences and
- Material Processing

The suitability of the Space Station for pointed experiments can be discussed with the following parameters

- altitude limitation
- orbit limitation
- mission duration
- attitude pointing error/stability
- data management & transfer
- power
- heat rejection
- cleanliness

The micro-g requirement is in the area of Life Sciences and Material Processing the dominant requirement. This results in special requirements to the altitude control/vibration isolation of the whole Space Station. No demand for pointing systems has been identified in these areas, thus they are no more considered in the following sections.

3.2 Astrophysics

The major objectives of astrophysics are

- investigate properties of extragalactic space, the milky way galaxy, and the solar system
- investigations with respect to cosmic evolution.

All wavelengths are used, like e.g.

- visible (cameras)
- IR-astronomy
- x-ray astronomy
- RF-astronomy

The driving requirements in the region of astrophysics are

- sensitivity (aperture, size, mass)
- pointing accuracies
- contamination limits
- thermal control (e.g. cryo systems etc.)

Typical payloads in the field of astrophysics are summarized in table 3.2-1 and table 3.2-2.

The manned Space Station can provide the following support for astrophysics experiments:

- manned operation
- manned maintenance & refueling of consumables
- contamination control
- perform special calibration procedures etc.

The location of an experiment at the Space Station will significantly enhance the overall utility (costs, operational mission efficiency).

	MASS (KG)	ALTITUDE (KM)	INCLINATION (DEG)	POWER (KW)	HEAT REJECTION (KW)	FIELD OF VIEW (DEG)	POINTING (ARCMIN)	STABILITY ARCSEC/ TIME	DATA RATE (MBPS)
SOT	8,200	400	57	6.8	0.9	0.025	0.017/90	0.1/15	50
SIRTF		100		1.3		0.125		2/20	
STARLAB		100		2.2		0.8		10/3	
SCRN		100		0.8		70		N/A	
SOLAR SOFT X-RAY TELESCOPE	1,300	430	57	0.2	0.2			0.1	
STO	16,6		57				8-12		
PINHOLE X-RAY CAMERA	10,0		97						
X-RAY OBSERVATORY	3,600	400	28.5	0.9	0.9		1.0		
HRS	1,800	400	< 45	0.5	0.5	10	6/90	36/0.02	0.03
XTE	1,000	400	28.5	0.6	0.6				
AXAF	10 TO 12,000	500	28.5	2.0	2.0		30	1.0	
LAMAR	9,500	400	28	3.4	0.4	1	3/67	10/0.02	0.1
VLBI	1,400	400	57	0.9	0.9	0.1	2.5/45	150/60	12
ASO	12,500	400	57	4.1		0.025	0.17/90	0.1/15	42

Mass (kg)

Inclination (deg)

Heat Rejection (kW)

Pointing (arcmin)

Data (mbps)

Altitude (km)

Power (kW)

Field of View (deg)

Stability sec/Time

Table 3.2-1: Typical astrophysical experiments I

Experiment Title	Pointing Accuracy	Major Dim. cm.	Mass kg	Power	Data
EUV Stellar Spectrometer	1' 30"/hr	60x40x20 30x20x20	20	20	
Large Proportional Counter	0.5 0.5	140x100x100	336	312	11M
Super Wide Angle Camera	0.3° 0.17°	500 x 130	30	100	
Heavy Ion Package (FSLP) ES 24, Beaujean	-	500 x 7	20	1	
Very Wide Field Camera (FSLP) ES 22, Courtès	5° 0.1°	100x100x50	90	50 150 pk	
High Resolution X-ray Background	0.5°	65x50x100	115	47 128 pk	3K
X-ray Sky Surveyor	0.5° 0.01°/sec	40x40x40	100	50	75K
UTEX	3' 3-10"/sec	1300 x 30 20x20	269 1	125 275 pk (300 thermal)	200K
Gas Scintilation Proportional Counter (FSLP) ES 23 ESA Andresen	1° 1°	30x30x55 18x22x10	20 2.8	12	42K
Wide Field X-ray Surveyor	1° 0.01°	240x110x110	100	35	100K
CIRBS	15' 15'	770 x 90 Pointing-sys.	90 30	58 100 pk	3K

Table 3.2-2: Typical astrophysical experiments II

Experiment Title	Pointing Accuracy	Major Dim. cm.	Mass kg	Power	Data
Large Area Detector Array	3' 0.5'/min				
Wide Field Camera for X-ray Bursts					
GIRL + Expt. package	10" 0.3"/sec	116.5 x 375	665	160	15K

Table 3.2-2 cont'd: Typical astrophysical experiments II

However, trade off studies will be required for the location if the contamination requirements will be very stringent. The payload may be preferred to be part of the space station, an unmanned space platform or a dedicated free-flyer. This trade off must include also manual versus automated operations aspects.

3.3 Earth and Planetary Remote Sensing

A set of typical payloads in the field of remote sensing is assembled in table 3.3-1.

The major objectives of the earth and planetary remote sensing are:

- Exploration of the solar system, incl. planets
- Earth dynamics, crustal motion, potential fields
- Resources Study
    - o crops
    - o minerals
    - o petroleum etc.
    - o ocean

The characteristics are

- Planetary landings (not relevant from pointing system view)
- Remote sensing
- Development of instruments for future missions.

The last two items are of interest for pointing system aspects. They have the following design driving requirements:

Experiment Title	Viewing Direction	Pointing Accuracy	Major Dim. [cm]	Mass [kg]	Power	Data
GIRL + Focal Plane Instr.	Limb	2' 1'/30 sec	116.50 x 375	605 (50)	340 w	16 Kbps
Ebert Fastie - GIRL + Focal Plane						
Optical Meso/Thermospheric Experiment	Limb	0.5° 0.01°/sec	20 x 60 x 15 15 x 60 x 15 35 x 60 x 15	19	80 w	8 Kbps
Grille Spectrometer (FSLP)	Limb	Exp. provided	180 x 60 x 80 (16.5 x 45 x 54)	80 (15)	300 w	50 Kbps
Lyman (FSLP)	Limb Nadir	S/L	71 x 16 x 25	8	35 w	200 bps
Microwave atmospheric Sounder	Limb	$\pm 2.5^{\circ}$ 0.01°/sec	30 x 30 x 50 NO 30 x 30 x 30 MRSE (10 x 30 x 10) AFD	11 (1)	40 w	3.4 Kps
Temp. + Wind in the Meso- + Thermosphere (FSLP)	Limb//vv	0.1° 0.05°/sec	52 x 59 x 81	37	52 w 70 wpk	2 Kbps
Waves in the OH Layer (FSLP)	Limb	S/L	19⌀ x 52.6	10.4	9 w 42 w	
Tropo/Stratospheric Wind	Limb	can provide own scan	100 x 100 x 60 (50 x 38 x 18)	25 (5)	8 w 12 w	10 Kbps
Na D-Line	Solar	1 - 3' 4"	183 x 24 x 25	12.5	30 w	ccTV

Table 3.3-1: Atmospheric and Solar Experiments

Experiment Title	Viewing Direction	Pointing Accuracy	Major Dim. [cm]	Mass [kg]	Power	Data
Solar Oblateness	Solar	1 - 5' 1 - 2"/sec	30ø x 100	20 (5)	25 w	9.6 Kbps
SCALP	Solar	2' 2^{o} 20" 0.5^{o}	30ø x 70	30	60 w	1.4 Kbps
Solar Spectrum	Solar	1.25^{o} 10'/15 min.	44 x 32 x 61	23	85 w 115 wpk	476 bps
Solar Constant 1 (FSLP)	Solar	$\pm 2^{o}$ 0.5^{o}/hr	16 x 16 x 44	5	3 w	
Solar Constant 2	Solar	$\pm 2.5^{o}$ ± 15	25ø x 0.5 22ø x 0.3 (23 x 48 x 35)	16 (7.5)	30 w 80 wpk	150 bps
Pallet mounted metric camera	Nadir	-	TBD	387	TBD	TBD
Scanning Radiometer	Nadir	S/L	25 x 14 x 30 (45 x 10 x 30)	15 (<10)	20 w	7 Kbps
Microwave Pressure Sounder	Nadir	S/L	50 x 50 x 20 30 x 60 x 20 50 x 25 x 25	31	400 w	100 bps
PICPAB (FSLP)	Earth	S/L	30 x 65.5 x 52.5 40 x 30 x 40 (44.7 x 30 x 40)	15+30 (25)	30 w	1 Mbps
MOMS	Nadir	0.1^{o} 0.01^{o}/sec	100 x 80 x 80 (single rack)	100 (65)	100 w	100 bps

Table 3.3-1 cont'd.: Atmospheric and Solar Experiments

Experiment Title	Viewing Direction	Pointing Accuracy	Major Dim. [cm]	Mass [kg]	Power	Data
Microwave remote sensing exp. (FSLP)	Earth	S/L	Antenna 200 x 125 x 180	152	1310 w 914 w 428 w	32 Mbp 377 Kbp
SAR	Earth $45^{o}/19^{o}$	S/L	Antenna 920 x 110 x 10 + Band Ant. 50 x 20 x 20 20 x 20 x 50	 150 (2)	2400 w 520 w 50 w	 57 Mbp
Low energy electron distribution function (FSLP)	Any	S/L	50 x 37 x 37	20 (1)	20 w	333 Kbp
Magnetic field vector (FSLP)	Any	S/L	11 x 15 x 45 14 x 14 x 16 Boom	5 10	7 w	256 bps

Table 3.3-1 cont'd.: Atmospheric and Solar Experiments

- Orbit
- Instruments pointing (optics, RF, etc.)
- data rates (RF-sensors)
- electrical power (LIDAR, Radars)
- RF-generation and susceptibility

For earth resource operational missions, where global coverage is required, highly inclined orbits (up to 90 deg) are required.

The payloads will comprise small to large microwave antennas and/or passive or active (large LIDAR) optical systems. Pointing requirements are today in the region of 1/10 of a degree, future large antennas may reduce this down to about 1/100 of a degree.

3.4 Environmental Observations

The major objectives of the environmental observations are

- atmospheric and ocean observation to further understanding of
  - o solar terrestrial interactions
  - o effects of man on environment
  - o effects of natural phenomena on environment
- Contribute to the development of global environmental models.

The major design drivers are

- global coverage (highly inclined orbit)
- broad spectral coverage (multisensor measurements)
- Cross Track Scanning, Viewing (multidirectional measurements)
- High data rates (up to 120 MBPS)
- very large antennas

The instruments may be single antennas or grouped on platform/pallets/bridges. The instruments comprise passive remote sensors, active stimulation by lasers, plasma wave injection facilities, electron beams and powerful radars. Typical characteristics for some payloads are shown in table 3.4-1. Early missions at low inclination may include missions for man supported equipment development missions.

The major design driving requirements are

- Orientation & pointing
- Data rates
- Power
- Orbit range.

	MASS (KG)	ALTITUDE (KM)	INCLINATION (DEG)	POWER (KW)	HEAT REJECTION (KW)	FIELD OF VIEW (DEG)	POINTING (ARC MIN)	STABILITY (ARC SEC/ TIME)	DATA RATE (MBPS)
OCEAN	10,000	400	57–90	10	10	1	720	720	120
LARS	1,200	780	> 60	1.7					50
UARS	2,400	400	56, 70	1.3	0.8	VARIOUS			0.02
SPACE PLASMA PHYSICS	3,200	3–400	57–90	2.7	1.8	45	60	60	7.5
ZERO-G CLOUD PHYSICS	500	ANY	ANY	1.4		N/A	N/A	N/A	0.5
METEOROLOGY	1,200	400	57	1.2	0.74		6	6	0.01
ICE AND CLIMATE EXPERIMENT	3,500	275	87	2.3					1.4 TO 17.0

TABLE 3.4-1: TYPICAL ENVIRONMENTAL OBSERVATION PAYLOADS

3.5 Communications

The space station mission of interest with respect to communications and pointing systems will be the technology development for advanced communications technology. The space stations large size, high prime power supply, availability of man to observe, and the recovery of the hardware makes it ideal to employ it as an in situ laboratory.

The major areas of interest are

- large deployable antennas
- Laser communications
- Space borne Interferometer
- millimeter wave propagation.

Larger antennas, Laser communications, interferometer etc. require all higher pointing performance than delivered by the Space Station itself.

3.6 Requirements Summary

3.6.1 Attitude Pointing/Stability/Mass

Astrophysics

Three major groups can be identified

1. Pointing accuracy — 1 arcmin down to arcsec, high stability requirements

   Most payloads are in the range from 200 to 8000 kg (only some are heavier)

2. Pointing accuracy — in the range of 0,1 deg

   Most payloads are in the range from 10 to 100 kg (only a few up to 270 kg)

3. Pointing accuracy — greater 1 deg

   Payload mass — 10 - 150 kg

Earth Remote Sensing/Environmental Observations

A broad spectrum of sensors is considered

- RF
- optical

with pointing requirements of 0.05 to 0.3 deg

Payload mass e.g.	LIDAR	up to 3500 kg
	RF, Optical	up to 150 - 200 kg
		(one payload up to 10000 kg, but only 720 arcmin pointing accuracy)

Communications

Large antenna pointing	$<< 0.1^{o}$
Major pointing constraints	low payload eigenfrequencies

3.6.2 Altitude/Orbit Limitations

- Altitude

  Nearly all missions can be satisfied by the 400 to 600 km circular orbit, only some earth viewing mission prefer altitudes up to 1000 km

- Inclination

  The inclination requirements can be summarized in three groups

  - o Astrophysics and low "g" prefer 28.5 deg inclination
  - o Earth viewing missions which can be satisfied by 57 deg
  - o Earth viewing missions requiring global coverage (i ≈ 90 deg)

3.6.3 Mission Duration

The potential benefit of the Space Station lies in the capability of supporting scientific research by man's presence of more than 7 to 30 days.

Most missions (e.g. Astrophysics, Earth observation) require mission duration in the region of months and years.

The useful life of the payloads can be increased by maintenance, replenishment of consumables and by the update or changeout of new technology equipment (e.g. smarter sensors etc.) thus increasing the utility of the observatories through longer on-orbit life.

3.6.4 Data Management & Transfer

High data rates will be required for

- astrophysics payloads (< 50 MBPS)
- environmental Instruments (< 120 MBPS) (e.g. RF equipments)

3.6.5 Power

The missions, related to the pointing system, with highest requirement for electrical power will be the LIDAR and RF missions with up to 25 kW.

Peak astrophysics requirements are 6.8 and 3.4 kW/payload. A variety of payloads of less than 1 kW power demand exists.

3.6.6 Heat Rejection

The requirements for heat rejection will ly in the same magnitude of the power demand. Special effort is required for RF and optical amplifiers active cooling.

3.6.7 Cleanliness

Contamination control for the sensor systems is a stringent requirement (e.g. IR-astronomy, x-ray astronomy). This may cause problems with a manned space station (e.g. local atmosphere cabin leakage or other sources), an accommodation of the affected payload on a space platform may be preferred.

3.6.8 Man Operated Functions

- Manned operation & resource provisoning of station-attached telescopes
- Assembly & checkout techniques
- In rare cases, even, it is conceivable that the investigator could actually be sent to the Space Station to perform his experiment
- Man conducted development of station mounted sensors, analytical & automated techniques
- Manned development of station-attached advanced systems (communications)

3.7 Space Station Constraints Summary

- Preferred space station orbit
  - o Altitude 400 - 500 km
  - o Inclination 28.5
- Space Station Characteristics
  - o Moments of Inertia >> Shuttle MOI
  - o Space Station 1. Eigenfrequency about 1 Hz (Shuttle/Pallet first eigenfrequency about 4 Hz)
  - o Local angular deflections at first eigenmode = large with respect to high pointing requirements
  - o Local disturbances due to actuators
man motion
RVD events
- Space Station attitude reference data can be delivered to pointing subsystems.

4. POINTING SYSTEM ACCOMMODATION ON THE SPACE STATION

4.1 Accommodation Analysis

The pointing stability requirements versus the mass of potential European Spacelab Experiments are summarized in Fig. 4.1-1. The mass/accuracy ranges of IPS and the PHM are indicated, a wide range of experiments can be covered by these two systems.

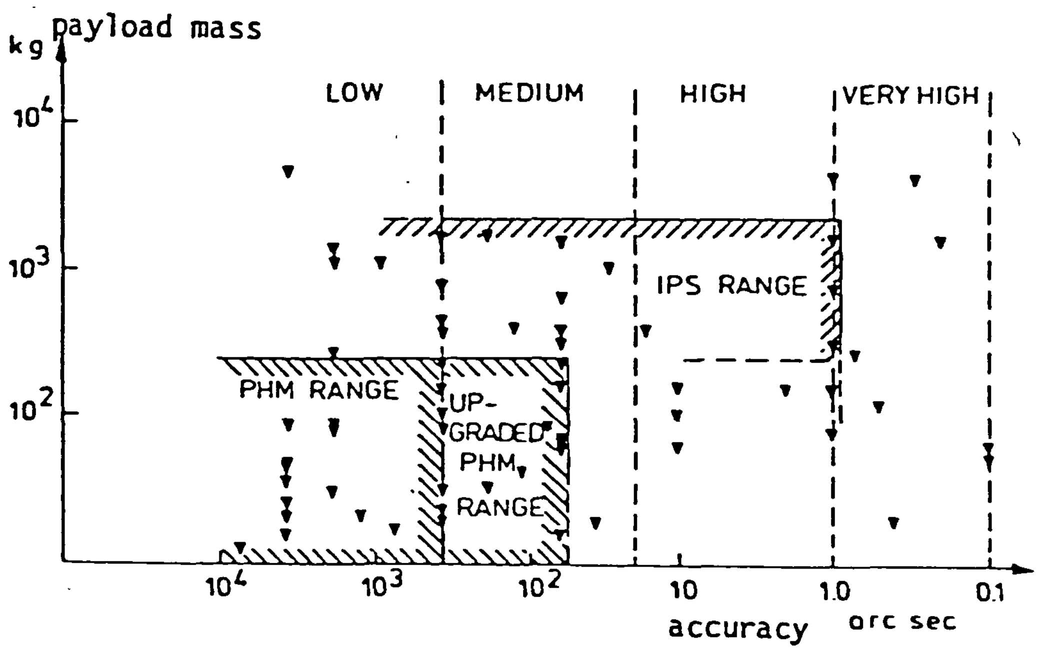


Fig. 4.1-1: IPS and PHM stability and payload range

4.1.1 IPS

Dynamics

- Orbiter

o Orbiter Limit cycle	± 0.1 deg/± 0.01 deg/sec
o Lowest Orbiter/Pallet Eigenfrequency	4 Hz
o Disturbances	Man motion Thruster firing

- Space Station

o Limit cycle	TBD
o Lowest Space Station Eigenfrequency	ca. 1 Hz (expected)
o Lowest eigenfrequency of Solar Array System	<< 0.1 Hz
o Disturbances	- Man motion - Distributed actuators (e.g. thrusters) - RVD activities - Moved parts (RMS etc.)

Rigid body angular accelerations of the Space Station due to disturbances and attitude control are expected to be lower in amplitude and frequency.

The disturbances are depending on

a) rigid body rates & angular accelerations

b) distance of IPS mounting location from Space Station C.O.M.

c) local translatorial accelerations and angular deflections due to Space Station flexibility.

to a)

Rigid body rates and angular accelerations are expected to be much lower than for the STS due to the Space Station high moments of inertia.

to b)

IPS performance simulations have been executed with a distance of about 1.6 m from C.O.M. For the Space Station a distance up to 10 - 15 m seems to be more realistic. Great attention has to be payed to the fact that resultant disturbances (lower rates, angular accelerations but much greater distances) are in compliance with the IPS-torquer capabilities (30 Nm).

to c)

Space Station flexibility

The first space station eigenmode of about 1 Hz requires at a first glance a lowering of the IPS bandwidth to less than 0.5 Hz. This is valid if the assumption can be made that the angular deflections due to the space station first eigenmode

are negligable. An IPS performance similar to the IPS/Orbiter configuration may be achieved, a quick analysis has shown that the IPS can handle a larger but slower disturbance better, than it can handle a smaller but faster disturbance.

If the local angular deflections of the first eigenmode have to be compensated by the IPS, the controller bandwidth has to be increased to 2 to 4 Hz, lying than within the Space Station structural frequencies. So the lower structural frequencies have to be notched in the controller.

The controller structure will be different to the existing one. Modifications which can improve the situation are e.g. decoupling (e.g. magnetic bearing) and control by inertial systems (Reaction wheels, CMG).

Much more investigations have to be performed for stability assessments. The Space Station FEM has to be used for detailed analysis. Interaction is also expected with the payload model. An adaptive controller is recommended due to space station and payload changing characteristics. The feedforward loop (accelerometers) is recommended not to be used.

Operations

Task sharing is performed between CDMS and DCU. For the Space Station an additional processor is recommended, it has the following advantages

- required for adaptive controller
- increase autonomy
- increase flexibility

Safety

Reduced safety requirements are expected, because no reentry is planned (no cargo bay door closing constraints).

Power and Data

- the payload support power can be upgraded according to future payload requirements.
- the payload data lines are according to RAU, CDMS, STS capabilities.

Payload mass

- the IPS has been designed for payloads from 200 to 7000 kg, this seems to fit also with most of the space station candidate payloads.

Improvements for IPS

- better Gyros (noise, drift)
- separate Sun-Sensor
- wide FOV Acquisition Sensor
- on-board alignment calibration

  between IPS and space station inertial measurement unit reduces initial IPS AMA attitude error which relieves from the wide FOV acquisition sensor after first acquisition after launch

- additional control-loop based not on gyros, but on gimbal-resolvers for pointing relative to Space Station (e.g. during Space Station rotations or during IPS stowage/deployment or parking, back-up mode for loss of gyros) → additional processor or RAM extension

- improvement of command-capability from the Experiment Computer (e.g. automatic sequencing)

- Improvement of bright star-triplet acquisition procedure (→ SW) for bright stars search

- sun-sensor as fast attitude-sensor for fast loop control and not for attitude determination filter (ADF)

  → ADF works only for roll-attitude and not for LOS in solar pointing

  → different AMA-concepts for

  $$\left.\begin{array}{l}\text{stellar}\\ \text{solar}\\ \text{earth}\end{array}\right\}\ \text{pointing}$$

- new/additional scan profiles

  o raster-point scan (stop-and-go)
  o sin/cos scan
  o etc.

- earth sensors in control-loop (landmark, horizon-sensors)

4.1.2 PHM

In general, the PHM is for hemispherical coverage for

- low to medium 2 axes pointing and stability requirements for
- small to medium sized payloads,

requiring from the Space Station in its non-autonomous operation mode the

- state vector of the Space Station to calculate a quasi-inertial attitude for inertial pointing or earth tracking.

Possible PHM users in the field of Astrophysics are smaller experiments running in parallel with advanced large astrophysical payloads who want to maintain independence and flexibility from those experiments.

Possible PHM users in the field of Environmental Observation are all kinds of antenna- or telescope-based experiments fitting the PHM capabilities.

The PHM can be upgraded without problems by use of dedicated sensors (Gyros, Optical Sensors). With the demonstration model a pointing accuracy of 0.5 arcmin was achieved with a Dornier off-the-shelf sun sensor.

No accommodation problems exist with payload power and data requirements.

The PHM controller bandwidth is nominally between 3 to 4 Hz. No interaction (as for the IPS) is expected between PHM and Space Station.

4.1.3 APM

Typical application would be in the fields of:

- the Space Stations own infrastructure such as TM/TC antennas for up-downlink purposes,
- antenna pointing for experiments with small, light weight antennas, and
- surveillance operations by supporting a (video) camera.

The APM can accommodate payloads with the performance and interface data as given in sect. 2.3.

4.2 Identification of Design Improvements

4.2.1 IPS

- Improvement of performance
  - o Adaptive/self optimizing control
  - o Modified controller/actuator concept
- Updated distributed microprocessor system
  - o more flexibility, more autonomy, intelligence distribution
- Technology improvements
  - o Sensor improvements, smart sensors (CCD/CID sensors etc.)
  - o decoupling from Space Station or carrier e.g. magnetic bearings

  o cryo or fluidic connections to the payload

- Improvements with respect to maintenance/operations.

4.2.2 PHM

- Accommodation of payload dedicated sensors (inertial, sun, earth reference)
- Development of standardized interfaces (mech. and data)
- Use of dedicated processor
- Increase slew rates

4.2.3 APM

- Accommodation of larger antennas
- more powerful motors to increase slew rates

4.3 Pointing Systems Accommodations Summary

Most of the considered payloads of section 3. can be accommodated by IPS, PHM and APM. Additional investigations primarily have to be done with respect to:

IPS

- IPS/Space Station dynamics
  - o definition of disturbances
  - o set up a coupled Space Station/IPS finite element model
  - o analyse modified controller concepts
  - o perform simulations

- IPS Processor Accommodation
  - o S/W Requirements
  - o Task sharing between DCU, new processor and CDMS

- Analysis of future sensor developments

PHM

- Accommodation of payload dedicated sensors
- Development of standardized interfaces

APM

- Analyse accommodation of larger antennas

All three systems seem to be very well suited to be used as standard equipment for future Space Station missions.

DORNIER SYSTEM

Participation in
NASA
Space Station Study

TITLE: EXTERNAL RADIATOR CONCEPT FOR THE SPACE STATION
TITEL: GRUMMAN DESIGN

DOCUMENT NO.: TN-SSS-DS-001
DOKUMENT NR:

ISSUE NO.: -
AUSGABE NR:

ISSUE DATE: 14.02.1983
AUSGABEDATUM:

PREPARED BY: Dr. Kreeb
BEARBEITET:

COMPANY: Dornier System GmbH
FIRMA:

AGREED BY: H. Preiß
GEPRÜFT:

COMPANY: Dornier System GmbH
FIRMA:

CONTRACT NO : -

PROJECT MANAGER
PROJEKTMANAGER

Technical Note

External radiator concept for the
Space Station
- Grumman Design -

1. Introduction

The radiator concept, described in this technical note, is based on the requirements and input data from Grumman (Datafax No. 6170 from 1/3/83) and is part of the cooperation between Grumman and Dornier System within the Space Station Study.

The note describes a version of radiator modules to be deployed, where each module is able to radiate up to about 1,1 KW (to one side only) and possesses a dimension of 2,5 m x 1 m.

The modules will be attached to a heater system .

Main requirements

- o Radiation energy : 25 KW
- o Radiator temp. limits : 280 K ÷ 340 K
- o Radiator modules attached to a freon 21 loop system
- o Radiator optimized with respect to weight, cost, redundancy and lifetime

The overall design consists of a number of these modules which possess the following main design features :

- o Heat dissipation on the modules by means of heat pipes.
- o Radiation to one side only.
- o For temperature control reasons the heat pipes on the module may be replaced by VCHP's (gas controlled heat pipes).

The following assumptions have been made concerning the first step of a heat rejection system with a maximum load of 25 KW which should be available and qualified in 1990.

<u>Assumptions</u>

- o Design based on state of the art technique
  - No development risk
  - Minimum manufacturing cost
  - Minimum development cost.
- o No welding during assembly in space.
- o Minimum temperature drop between freon 21 loop and radiator surface.

The deployable radiator is in competition with a modular design where the (heat-pipe)-radiator modules will be attached to the cooling loop (the radiator modules will have a similar design but another coupling mechanism). The main advantage of deployable modules is a smaller temperature drop and an easier and more reliable accommodation technique. The design goal is therefore a good thermal coupling to the loop system.

2. Design Aspects

The most promising concept concerning radiator systems of about 25 KW_{th} performance is a central freon loop (already realized in Shuttle and Spacelab and foreseen for the European Eureca) and attached heat pipe radiator modules either as deployable parts or as modules to be assembled in space. Depending on the design configuration, the central freon 21- loop may possess in some areas flexible ducts which do not seem to be a problem.

A major limitation with all these concepts is that the total system is lost if the central tube is punctured. A tube puncture is possibly of acceptably low probability for the length of a Shuttle mission, but it becomes a very high probability in space stations planned for a mission duration of 10 years.

This loop must therefore be designed in a redundant way or it must be protected thoroughly against micrometeoroid penetrations. The heat transport and distribution in the individual radiator modules will be done by means of heat pipes which consist of individually sealed tubes. The damage of one or a couple of heat pipes will therefore not influence the energy radiation dramatically.

Furthermore, the modules and especially their deployment mechanism should be designed in such a way that the (feeder) heat pipes of the module can be coupled either to a central loop system as mentioned or to another heat pipe which acts as a header. Since such a header heat pipe would supply heat to a number of feeder heat pipes, the header would obviously have to be of greater heat transport capacity.

There exists the possibility of providing some or all radiator modules with variable conductance heat pipes (VCHP's) instead of with single fix-conductance heat pipes in order to provide self-regulating control of the radiator temperature.

If necessary these heat pipes can be provided with an active control mechanism. All these heat pipe control techniques have already been developed, tested and space-proved, therefore they are not a major design point.

The major problem within the deployable radiator concept is that the module must be deployable (the moveable, thermally conducting joint).

In principle, the following solutions must be discussed :

- flexible header heat pipe
- collapsible hinges
- roll contacts
- eccentric tubes
- slip rings
- concentric tubes, interface filler
- flexible metals (eg. copper wire)
- radial heat pipe.

In an ESTEC study (Contract No. 3314/77/NL/PP (SC)) Dornier System has discussed and evaluated all these solutions except the one involving a flexible heat pipe. We do not recommend a flexible feeder heat pipe because such a heat pipe has under zero-g a relatively low performance, higher corrosion risk, higher risk of developing a leak and of degradation due to micrometeoroid penetrations.

If the problem of the deploy mechanism concerning the heat transport can be solved, this radiator concept imposes no major restriction.

Preferred deploy configuration

Because of the temperature level 280 K - 340 K, the reliability and required performance, the heat pipes of the deployable radiators and the mounting platform will be of an aluminium/ammonia type. Figs. 1 and 2 show some profiles which are available at Dornier System. It is no problem to supply new profiles with an outer shape which is more convenient for the envisaged design.

If the radiator modules are to be deployed once only, a design such as the one shown in Figs. 3 and 4 may be advantageous. If the modules are to be deployed several times or if they have to be moved continuously, a design such as the one shown in Figs. 5 and 6 may be preferred.

Under ESTEC Contracts (3314/77/NL/PP and 3765/78/NL/PP) we developed and tested deployable radiators (lab models) according to Figs. 5 and 6. These studies included a pre-selection of the concepts where the coupling length to the header was 1 m and the corresponding inner diameter 15 mm.

The following table contains the results obtained :

technique :	heat resistance (K/W) :
collapsible hinges	3
roll contacts	0.8 - 1.6
eccentric tubes	0.7 - 1.4
slip rings	0.5 - 1.0
concentric tubes	
filler: grease, $\alpha = 3000\ W/m^2K$	0.02 - 0.03
filler: 25% grease, 75% Ag-powder	
$\alpha = 3000\ W/m^2K$	0.01 to
	$4.5 \cdot 10^{-4}$
filler: mercury	0.01
flexible copper wire	0.07
radial heat pipe	0.025

The technique with the grease plus silver powder was carried out and tested in 2 configurations (Fig. 5 of configuration 1 and Fig. 6 of configuration 2).

For configuration 2 a special header/heat pipe profile was developed (DS-WR 15 in Fig.2) and the corresponding feeder heat pipes of the radiator module were welded in T-bar configuration to this header heat pipe (Fig. 6,7).

As a result it can be said that a heat transfer coefficient of 2500 W/m^2K and a torque of 1Nm can be reached in the deploy mechanism (length 1 m; diameter 15 mm).

3. Module Design

The lay-out is based on a maximum radiated heat of 450 W/m^2 (no sun on the panels, 330 K radiator temp.) and a minimum radiated heat of 150 W/m^2 (50% of the surface area in sunlight, 280 K radiator temp). The radiator efficiency is included and it is assumed that only one side of the radiator is able to radiate, the back side will be insulated. This leads to an average radiation heat of about 300 W/m^2 and a total radiation area of 84 m^2 for 25 KW.

We assume a coupling length to the header of 1 m for one radiator module and a module length of 2,5 m. This results in 34 modules with a coupling length of about 35 m (length of the loop system in the radiator area).

The minimum radiation heat per panel will be 1,125 KW. This value leads to a heat pipe performance of 380 W per heat pipe or 500 W x m if the panel possesses three heat pipes only (Fig. 8). Another design (Fig. 9) possesses 6 heat pipes where one heat pipe must possess a performance of 250 W x m.

The second design is compatible with the deploy mechanism, sketched in Fig. 5 where the heat pipes are welded with their integrated fin to a rotating structure. A performance of about 250 Wm has already been reached and qualified with aluminium/ammonia heat pipes which possess open axial grooves as capillaries eg. the profile DS-WR 10 (Fig. 2). The weight of the profile is about 0,230 Kg/m (without integrated fin) so that one panel will possess a heat pipe weight of 4 Kg (no gas reservoirs). In this case, the panel itself will consist of a single aluminium plate, bonded to the heat pipes; a honeycomb structure will not be necessary because of the stiffness of the heat pipe profiles. The total weight of such a radiator module will amount to 12 Kg.

The radial heat flux to the heat pipes will be about 2,5 W/m^2, a value which indicates no problem for ammonia heat pipes. The corresponding temperature drop between central loop and heat pipe vapour was calculated to 10 K. This relatively heavy design has the advantage of a high redundancy because of the large number of parallel heat pipes on one module. The loss of one heat pipe will result in a slightly higher temperature drop but it will not cause a dramatic change in the energy radiated.

The second design according to Fig. 6 possesses 3 heat pipes in the radiation sheet of one module only and these heat pipes are connected via a T-bar (Figs. 6 and 7) with a heat pipe which is part of the header system. This design has already been realized and tested under ESTEC Contract No. 3765/78/NL/PP . The header heat pipe profile is shown in Fig. 2 (profile DS-WR 15). Because of the T-bar connection, this design consists of one heat pipe system per panel only and it has therefore the disadvantage of a lower redundancy but the advantage of a very small temperature drop between the header system and the heat pipes of each module.

4. Temperature Control

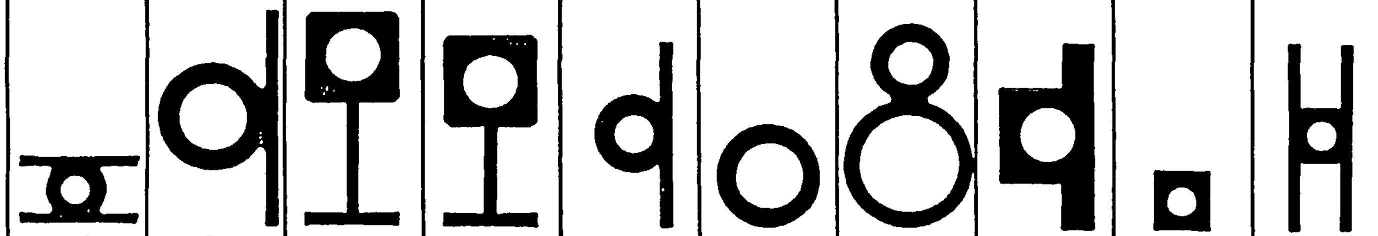
Without any great effort in designing and manufacturing, such a radiator module may be provided with gas-stabilized heat pipes (VCHP's) so that these modules may serve not only for the radiation but also for the temperature stabilization without any active electrical heater or controller systems. However, VCHP-provided modules would increase the weight considerably, so it has to be discussed whether it is sufficient to provide only some of the radiator modules with a VHCP-temperature stabilisation system. Such a lay-out can be performed as soon as more confirmation concerning the orbit data and radiation from sun and earth to the radiator are available.

5. Mechanisms

TBD.

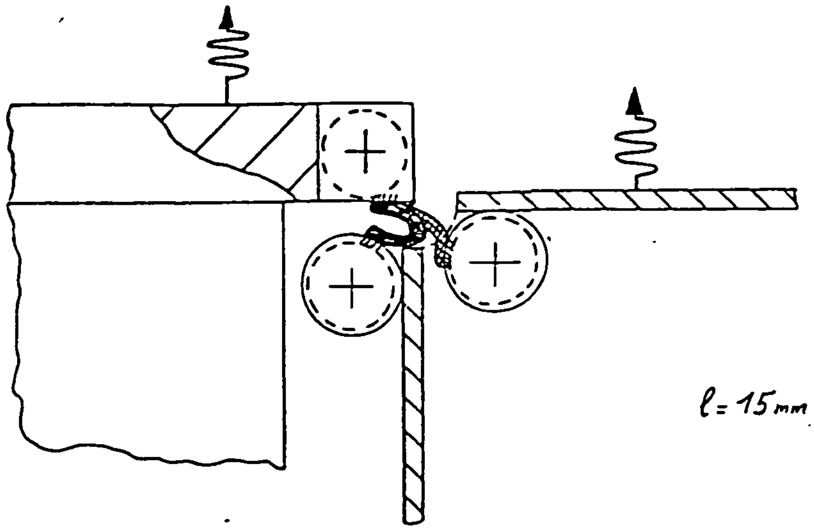
	 WR 1a	 WR 1b	 WR 1c	 WR 2	 WR 3	 WR 4	 WR 5	 WR 6	 WR 7	 WR 8
Rillentiefe A [mm]	0,7	0,9	1,2	0,9	0,9	0,9	0,9	0,8	0,8	1,0
Rillenbreite B [mm]	0,5	0,7	0,9	0,7	0,7	0,7	0,7	0,5	0,5	0,5
Rillenanzahl NR	24	18	12	30	30	18	30	20	20	20
Innendurchmesser [mm]	8,6	8,2	7,6	13,2	16,2 x 4,2	8,2	16,2 x 4,2	5,4	5,4	5,0
Außendurchmesser [mm]	12,0	12,0	12,0	18,0	20,0 x 8,0	12,0	20,0 x 8,0	12,0 x 12,0	12,0	9,0
Findicke [mm]	–	–	–	–	–	1,5	2,0	–	1,5	1,5
Finbreite [mm]	–	–	–	–	–	30,0	40,0	–	30,0	30,0
Profilmasse [kg/m]	0,122	0,129	0,140	0,266	0,200	0,235	0,309	0,300	0,333	0,200
Minimale Bauhöhe [mm]	12,0	12,0	12,0	18,0	8,0	12,5	8,0	12,0	12,0	9,5

Fig. 1: **Dornier System Rillen - Wärmerohr - Profile**



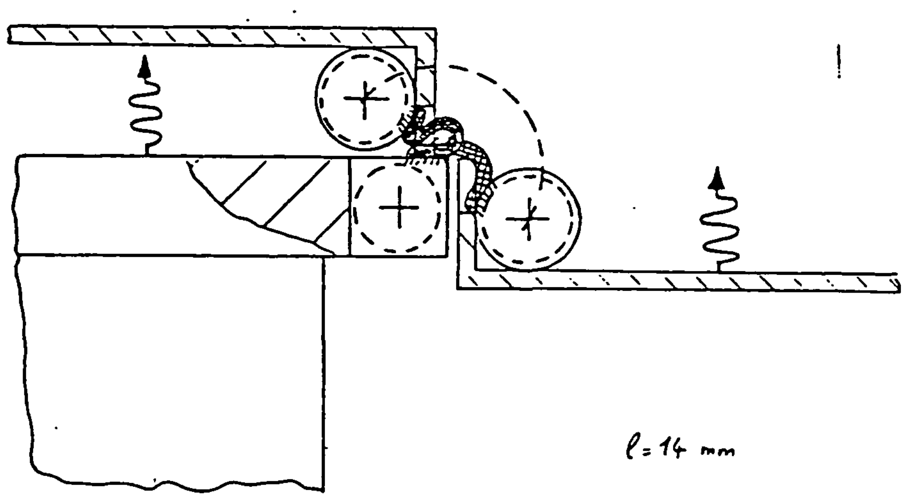
	WR 9	WR 10	WR 11	WR 12	WR 13	WR 14	WR 15	WR 16	WR 17	WR 18
Grooves depth A [mm]	1,0	1,4	1,4	1,4	1,5	1,5	1,5	1,4	1,0	1,0
Grooves width B [mm]	0,5	0,8	0,6	0,6	0,7	1,0	0,7	0,6	0,5	0,5
No. of Grooves NR	20	32	32	32	22	24	22	32	20	20
Inner diameter [mm]	5,0	11,2	9,2	9,2	7,0	10,6	7,0	9,2	5,0	5,0
Outer diameter [mm]	9,0	17,0	15,5×15	15,5×15	12,0	16,0	12,0	15,0×30,0	9,0×9,5	9,0
Fin thickness [mm]	1,5	2,0	2,0	2,0	1,5	–	–	5,0	–	1,8
Fin width [mm]	30,0	35,0	15,0	15,0	30,0	–	–	30,0	–	30,0
Mass [kg/m]	0,335	0,335	0,604	0,572	0,250	0,210	0,465	0,5	0,150	0,34
Min height [mm]	10,0	17,5	35,0	31,0	12,5	16,0	32,2	16,0	9,0	9,5

Fig. 2 DORNIER SYSTEM AXIAL GROOVED HEAT PIPE PROFILES



90° - turn

Fig. 3



180° - turn

Fig. 4

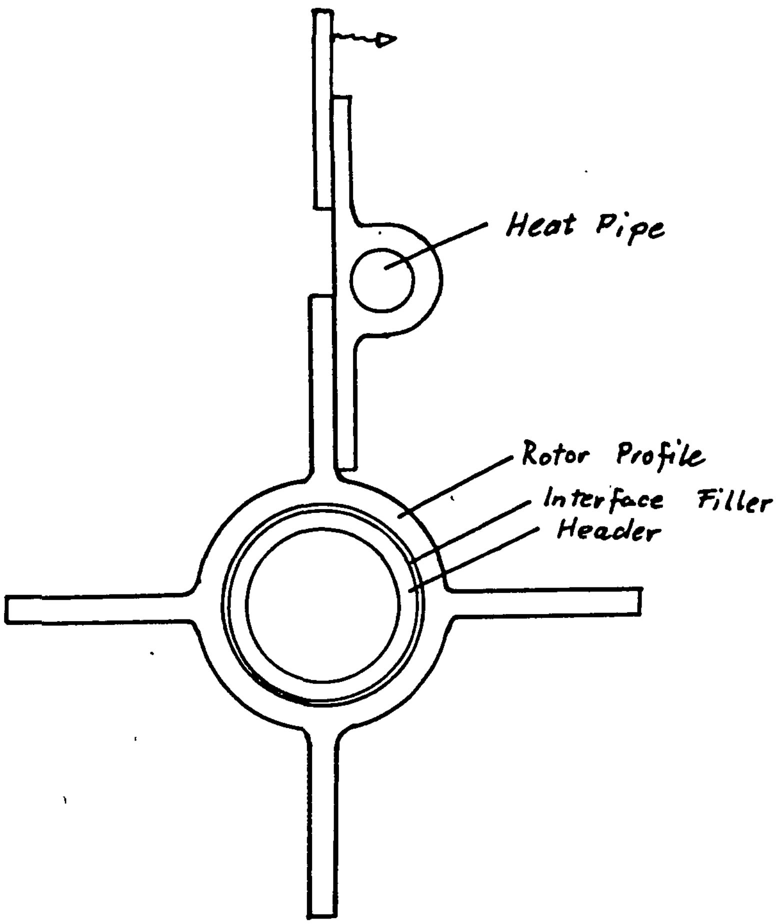
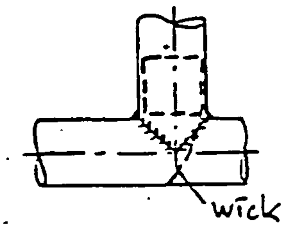
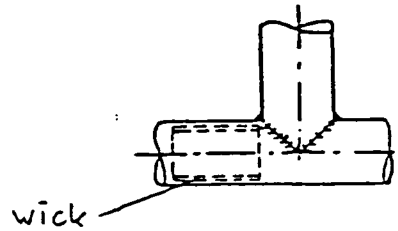


Fig. 5 Configuration 1

version c_1:



version c_2:



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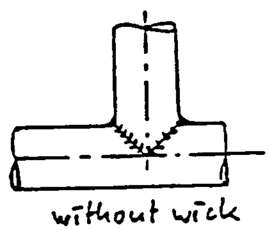

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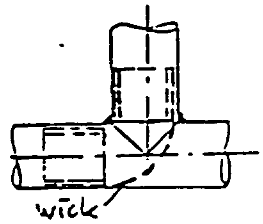


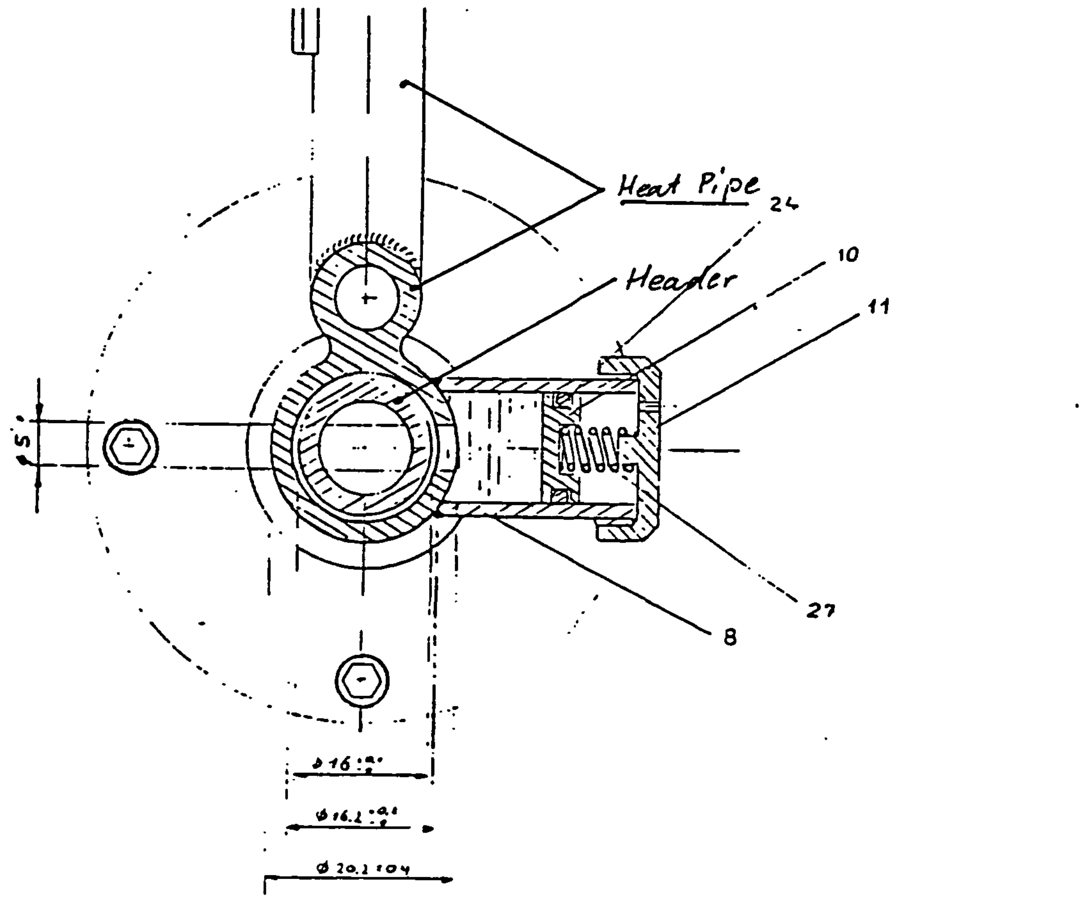


Fig. 6: Replenishing Valve and T-bar Heat Pipe (configuration 2)

DORNIER SYSTEM

Participation in NASA Space Station Study

-14-

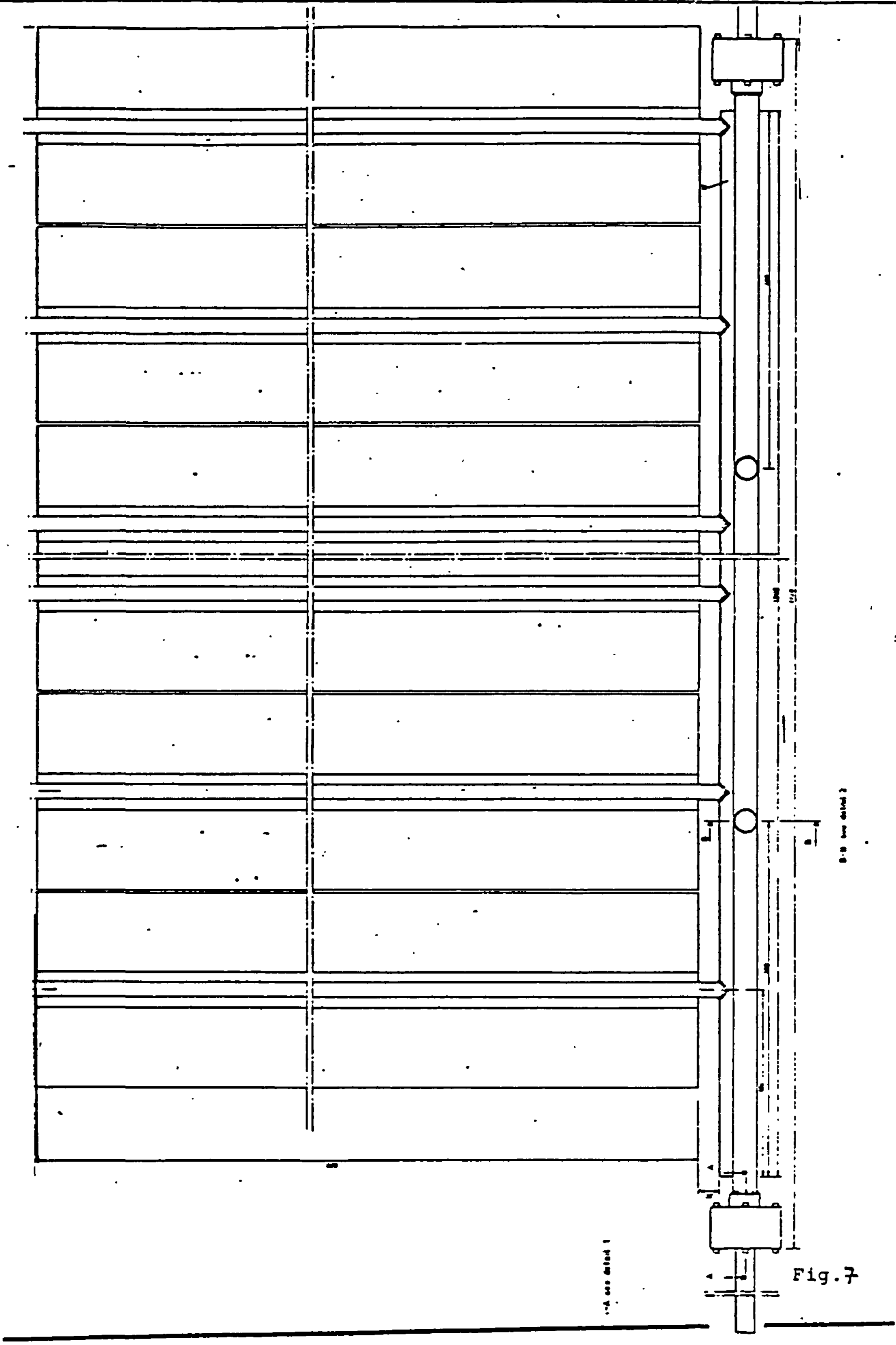


Fig.7

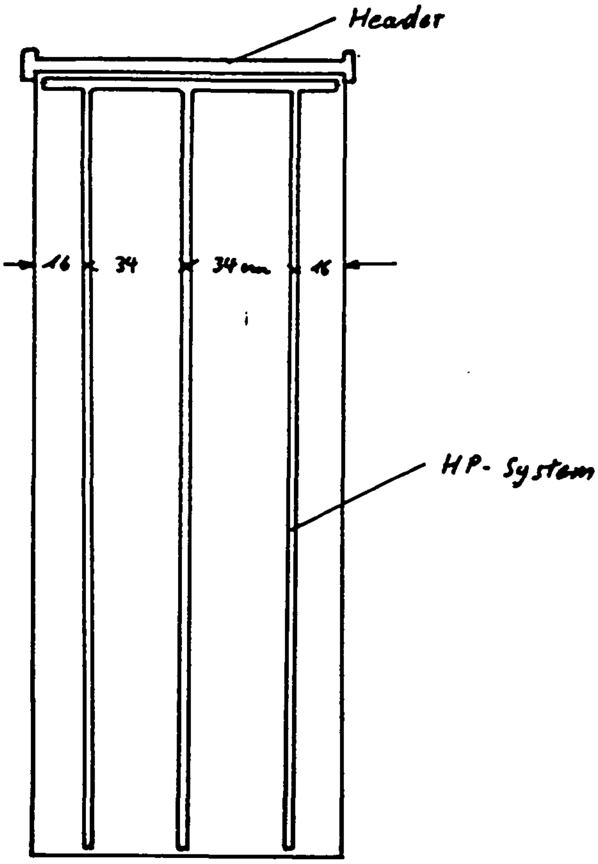

Header
16
34
34 mm
16
HP-System

Fig. 8

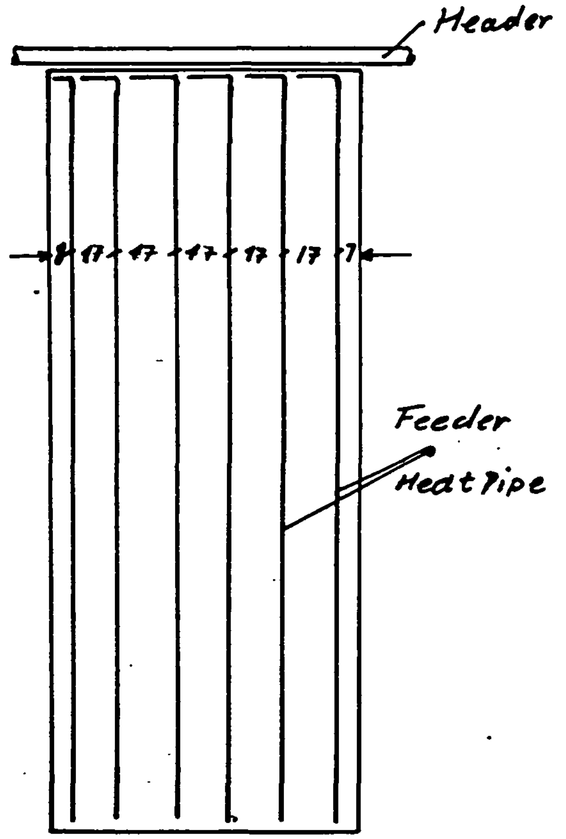

Header
8
17
17
17
17
17
7
Feeder
Heat Pipe

Fig. 9

BRITISH AEROSPACE P.L.C.

ROLL-UP SOLAR ARRAYS

Future Shuttle-launched spacecraft will require high power deployable/retractable solar arrays. The space Telescope solar Array for which British Aerospace is the prime contractor forms the basis for a family of arrays to power a wide variety of Shuttle-launched and other missions.

GENERAL DESCRIPTION

The array (as developed for the space Telescope) comprises two interchangeable wings, each employing a double roll-out flexible solar cell blanket.

The array delivers 4.2 kW of power at 34 V after two years.

The principal design features are:

- Automatic or manual deployment and retraction in orbit to permit satellite reboost or return to earth
- Independent or synchronous orientation of the wings
- Independent operation of Secondary and Primary Deployment Mechanism on each wing
- On-blanket shunt diode protection of all cells against shadowing
- Separately mounted blocking diodes for cell isolation during eclipses, and also in the event of short circuits
- Fully redundant drive electronics and drive motors
- Unique capability of on-orbit jettison or replacement
- Heater protection for critical low temperature items
- Very low interactive torques imparted to the attached satellite during array rotation (using the STSA Solar Array Drive)
- Qualification to 30,000 thermal cycles (corresponding to approximately five years in low earth orbit)
- Space qualified by the end of 1981.

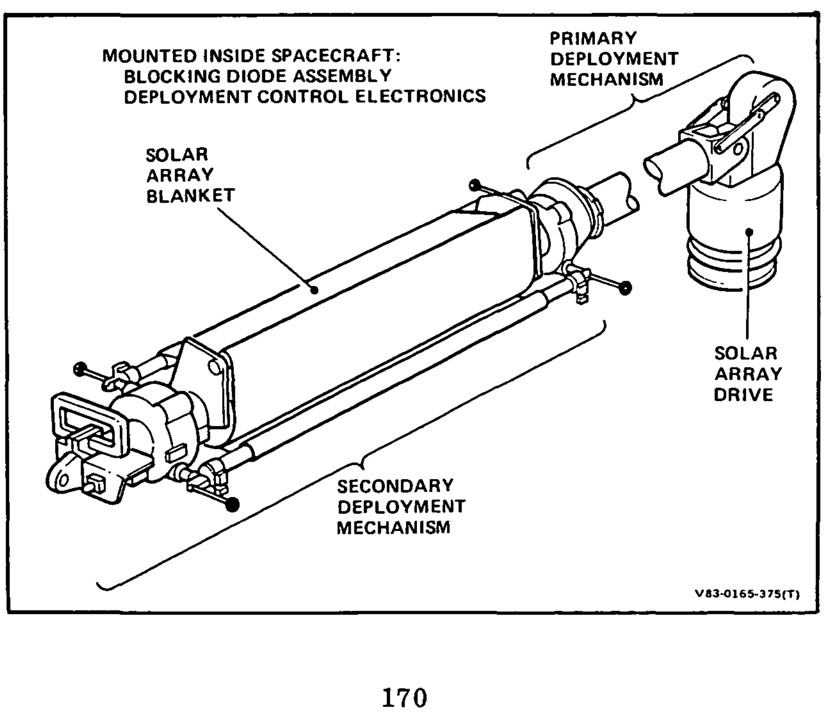
A schematic diagram of a stowed wing is shown over.

It should be noted that the Solar Array Drive is likely to be mission specific, and the design of the Primary Deployment Mechanism may be varied to suit user's individual requirements.

CHARACTERISTICS

OUTPUT AT 34V (ARRAY)	4.8KW at beginning of life 4.2KW after 2 years in orbit — at 90° (Arranged in 20 Solar Panel Assemblies, or SPAs)
ARRAY MASS (TWO WINGS)	Solar Cell Blankets (4) — 86.2 Kg Secondary Deployment Mechanisms (2) — 111.8 Kg Blocking Diode Assemblies (2) — 14.9 Kg Deployment Control Electronics (1) — 8.5 Kg MISSION INDEPENDENT MASS — 221.4 Kg Primary Deployment Mechanisms and Jettison Adaptors (2) — 37.8 Kg Primary electrical harnesses (2) — 10.6 Kg MISSION DEPENDENT MASS (STSA) — 48.4 Kg
STOWED DIMENSIONS (ONE WING)	4.36m x 0.65m x 0.70m (including PDM and SADM)
DEPLOYED DIMENSIONS (ONE WING)	Width 2.83m Length 11.82m (excluding PDM and SADM)
NOMINAL LIFETIME	5 years (30,000 thermal cycles)
INTERFACES (ONE WING)	- RMS grapple point for array jettison/replacement - EVA access/overrides for all mechanisms - Adaptor/Jettison interface - Primary Deployment Mechanism/Solar Array - Drive Interface - Stowage latches (2)

V83-0165-376(T)



GRUMMAN

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